FOR ERRATA

AD________

410 255

THE FOLLOWING PAGES ARE CHANGE

TO BASIC DOCUMENT

AD 410 255

END ITEM CONFIGURATION CHART

Specification Issue	ECP'S	Production Effectivity
Basic 1 July 1963	MA5-3	NA225173 and subsequent
	MA5-5	
	MA5-8	
	MA5-11	
	MA5-16	
	MA5-23	
	MA5-31	
	MA5-38	
	MA5-39	
	MA5-42	
	MA5-55	
	MA5-59	
	MA5-62	
	MA5-65	
	MA5-70	
	MA5-71	
	MA5-72	
	MA5-76	
	MA5-77	
	MA5-80	
	MA5-84	

The YLR105-NA-7 (Sustainer) engine defined by this specification incorporates the above listed engineering changes which give it a baseline configuration of YLR105-NA-7 MD 5x7x11x13 15x17x20 23x25 28x31.

FORM 608-B PLATE REV. 1-58

SUPPLEMENTARY INFORMATION

YLR105-NA-7 MD Ident. No.	MA5- ECP No.	Description
1	none	Replacement of Head Suppression and Propellant Utilization Valve Assemblies
2	none	Changes to Pneumatic Start System Control Assembly
3	none	Improvement of Lube and Fuel Drainage Provisions
4	none	Replacement of Thrust Chamber Injector-Dome Flush Nozzle
5	none	Installation of Tapered Inducer on Turbopump Assembly
7	3	Redesign of Fuel Pump Elbow Seal on Turbopump Assembly
11	5	Replacement of Mixture Ratio Control Assembly
13	31R2	Addition of Vernier Thrust Chamber Hypergolic (Pyrophoric) Ignition System
14	39R2	Addition of Fuel Pressure Sensing Switch
15	42	Redesign of Fuel Manifold Start System Pressure Switches
17	55	Replacement of Studs on Fuel Start Tank and Vernier Feed Tank
20	62R1	Replacement of Engine Electrical Relays on Interconnecting Box
21	11R2 & 72R1	Installation of Special Purpose Vernier Diodes on Interconnecting Box; and Changes to Engine Relay Box Redundant Head Suppression Solenoid Control Circuit
22	76R2	Changes to Pneumatic Start System Control Assembly
23	59R3	Replacement of Oxidizer Vent and High Pressure Relief Valve Assembly
25	77	Redesign of LOX Bootstrap Scoop
26	70R1	Replacement of Lube Tank Pressurizing Valve Assembly
27	80R1	Incorporation of Kel-F Liner Oxidizer Pump Inlet Adapter
28	65 & 71	Redesign of Lube Oil Drain System; and, Turbine Exhaust Aspirator
31	84	Replacement of Aluminum "B" Nuts
none	8	Reidentification of Accumulator
none	16	Revise Packaging Requirements
none	23R1	Changes to Purity (Contamination) Requirements
none	38R3	Incorporation of Furnace Brazed Thrust Chamber

3.5 <u>Drawings and data</u>.- The following Rocketdyne drawings and data shall form a part of this specification:

(a)	Engine assembly (sustainer)	1002001
(b)	Markings installation	1002004
(c)	Control system installation (gas generator system)	3002001
(d)	Propellant feed system installation	4002001
(e)	Lubrication system installation	35020[illegible]
(f)	Electrical system installation	5002001
(g)	Engine loose equipment	6502001
(h)	Expendable loose equipment	6502002
(i)	Pyrotechnic loose equipment	6502003
(j)	Hydraulic system installation	6002001
(k)	Exhaust system installation	3002010

<u>Model identification numbers</u>

<u>ENGINE AND MPL</u>	<u>MD NUMBERS</u>
YLR105-NA-7	5x7x<u>11x13</u> <u>15x17x20</u> <u>23x25</u> <u>28x31</u>

3.5.1 <u>Before contract</u>.- Not applicable.

3.5.2 <u>After contract</u>.- Not applicable.

N-63-4-3

ROCKETDYNE

A DIVISION OF NORTH AMERICAN AVIATION, INC.

CANOGA PARK, CALIFORNIA

R-3952S

MODEL SPECIFICATION
LIQUID PROPELLANT ROCKET ENGINE
ROCKETDYNE MODEL YLR105-NA-7
(SUSTAINER)

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.
6633 CANOGA AVENUE
CANOGA PARK, CALIFORNIA

(MA-5 Block I)

PREPARED BY

R. B. Scott
Contract Specifications

APPROVED BY

W. J. Brennan

W. J. Brennan
Chief Engineer
Liquid Rocket Engineering

NO. OF PAGES 99 & ix

REVISIONS

DATE 1 July 1963

DATE	REV. BY	PAGES AFFECTED	REMARKS

FORM R 10-G (PLATE)

END ITEM CONFIGURATION CHART

Specification Issue	ECP's	Production Effectivity
Basic 1 July 1963	MA5-3 MA5-5 MA5-8 MA5-11 MA5-16 MA5-23 MA5-31 MA5-38 MA5-39 MA5-42 MA5-55 MA5-59 MA5-62 MA5-65 MA5-70 MA5-71 MA5-72 MA5-76 MA5-77 MA5-80 MA5-84	NA225173 and subsequent

The YLR105-NA-7 (Sustainer) engine defined by this specification incorporates the above listed engineering changes which give it a baseline configuration of YLR105-NA-7 MD7x11 15x17x20 23x25 28x31.

SUPPLEMENTARY INFORMATION

YLR89-NA-7 MD Ident. No.	MA5- ECP No.	Description
1	none	Replacement of Head Suppression and Propellant Utilization Valve Assemblies
2	none	Changes to Pneumatic Start System Control Assembly
3	none	Improvement of Lube and Fuel Drainage Provisions
4	none	Replacement of Thrust Chamber Injector-Dome Flush Nozzle
5	none	Installation of Tapered Inducer on Turbopump Assembly
6	11R2	Installation of Special Purpose Vernier Diodes on Interconnecting Box
7	3	Redesign of Fuel Pump Elbow Seal on Turbopump Assembly
11	5	Replacement of Mixture Ratio Control Assembly
13	31R2	Addition of Vernier Thrust Chamber Hypergolic (Pyrophoric) Ignition System
14	39R2	Addition of Fuel Pressure Sensing Switch
15	42	Redesign of Fuel Manifold Start System Pressure Switches
17	55	Replacement of Studs on Fuel Start Tank and Vernier Feed Tank
20	62R1	Replacement of Engine Electrical Relays on Interconnecting Box
21	72R1	Changes to Engine Relay Box Redundant Head Suppression Solenoid Control Circuit
22	76R2	Changes to Pneumatic Start System Control Assembly
23	59R3	Replacement of Oxidizer Vent and High Pressure Relief Valve Assembly
25	77	Redesign of LOX Bootstrap Scoop
26	70R1	Replacement of Lube Tank Pressurizing Valve Assembly
27	80R1	Incorporation of Kel-F Liner Oxidizer Pump Inlet Adapter
28	65 & 71	Redesign of Lube Oil Drain System; and, Turbine Exhaust Aspirator
31	84	Replacement of Aluminum "B" Nuts
none	8	Reidentification of Accumulator
none	16	Revise Packaging Requirements
none	23R1	Changes to Purity (Contamination) Requirements
none	38R3	Incorporation of Furnace Brazed Thrust Chamber

SPECIFICATION CHANGE INDEX

Specification No. R-3952S

SCN No.	MA5 ECP No.	SCN Date	Pages Affected	Item Affected YLR105-NA-7 MD Ident. No.
1 *	various *	1 July 1963	various *	various *

* NOTE: SCN 1 to R-3952S incorporates the sustainer portion of the following approved SCN's to R-2850aS:

R-2850aS SCN No.	MA5 ECP No.	R-2850aS SCN Date	Paragraphs Affected	Item Affected YLR105-NA-7 MD Ident. No.
2	68	24 July 1962	3.5, 3.5.3	24
11	73R1	5 December 1962	3.5	30

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.

SPECIFICATION CHANGE NOTICE

Number 1
(ref: R-2850aS/SCN 11)*

SECURITY CLASSIFICATION: UNCLASSIFIED [X] CONFIDENTIAL [ ] SECRET [ ]

Date 1 July 1963

SUPERSEDES Date

1. FOR ECP NO.	2. NOMENCLATURE AND MODEL	3. SPECIFICATION NO.
NA-MA5-73R1(ref)*	Rocketdyne Liquid-Propellant Rocket Engine Model YLR105-NA-7 (Sustainer)	R-3952S
4. CONTRACT	**5. CONTRACTUAL AUTHORIZATION**	**FILE OPPOSITE SPEC. PAGE NO.**
AF04(694)-58	CCN40(N58-874)*	17

6. PRODUCTION EFFECTIVITY:

NA 225181 and subsequent

7. EFFECT OF CHANGE ON SPECIFICATION CONTENT:

Paragraph 3.5 Drawings and data

Change YLR105-NA-7 MD number identification to incorporate MD30.

* NOTE: Previously proposed and authorized as noted.

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.

SPECIFICATION CHANGE NOTICE

Number 1
(ref: R-2850aS/SCN 2) *

Page 2 of 4

SECURITY CLASSIFICATION
- UNCLASSIFIED [X]
- CONFIDENTIAL [ ]
- SECRET [ ]

Date 1 July 1963

SUPERSEDES Date

1. FOR ECP NO.	2. NOMENCLATURE AND MODEL	3. SPECIFICATION NO.
NA-MA5-68 (ref)*	Rocketdyne Liquid-Propellant Rocket Engine Model YLR105-NA-7 (Sustainer)	R-3952S
4. CONTRACT	5. CONTRACTUAL AUTHORIZATION	FILE OPPOSITE SPEC. PAGE NO. 17
AF04(694)-58	CCN 24 (N58-315)*	

6. PRODUCTION EFFECTIVITY:

NA225192 and subsequent.

7. EFFECT OF CHANGE ON SPECIFICATION CONTENT:

Paragraph 3.5 Drawings and data

Change YLR105-NA-7 MD number identification to incorporate MD24.

* NOTE: Previously proposed and authorized as noted.

Form 608-B-8

ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC.

SPECIFICATION CHANGE NOTICE

Number 1 (ref: R-2850aS/SCN 2)*

Page 3 of 4

Date 1 July 1963

SUPERSEDES Date

SECURITY CLASSIFICATION: UNCLASSIFIED [X] CONFIDENTIAL [ ] SECRET [ ]

1. FOR ECP NO.	2. NOMENCLATURE AND MODEL	3. SPECIFICATION NO.
NA-MA5-68 (ref)*	Rocketdyne Liquid-Propellant Rocket Engine Model YLR105-NA-7 (Sustainer)	R-3952S
4. CONTRACT	5. CONTRACTUAL AUTHORIZATION	FILE OPPOSITE SPEC. PAGE NO. 18
AF04(694)-58	CCN24 (N58-315)*	

6. PRODUCTION EFFECTIVITY:

NA225192 and subsequent.

7. EFFECT OF CHANGE ON SPECIFICATION CONTENT:

<u>Paragraph 3.5.3 Weights</u>

Change the paragraph to read as follows:

<u>Weights</u>.- The dry weight of the YLR105-NA-7 shall not exceed 1028 pounds. The wet weight after normal shutdown shall not exceed 1152 pounds. The wet weight with all systems filled to capacity shall not exceed 1417 pounds.

Assembly	Estimated Weight, lb
Thrust chamber	367
Mount, gimbal assembly	18
Turbopump installation	229
Oxidizer system	46
Fuel system	40
Gas generator system	45
Lubrication system	21
Electrical system	23
Pneumatic system	7
Exhaust system	85
Hydraulic system	45
Ignition system	6
Dry weight total	932
Wet weight total	1086
Wet weight burnout	1024

Form 608-B-8

* NOTE: Previously proposed and authorized as noted.

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SPECIFICATION CHANGE NOTICE

Number 1
(ref: R-2850aS/SCN 2)*

Date 1 July 1963

SECURITY CLASSIFICATION: UNCLASSIFIED [X] CONFIDENTIAL [ ] SECRET [ ]

SUPERSEDES Date

1. FOR ECP NO.	2. NOMENCLATURE AND MODEL	3. SPECIFICATION NO.
NA-MA5-68 (ref)*	Rocketdyne Liquid-Propellant Rocket Engine Model YLR105-NA-7 (Sustainer)	R-3952S
4. CONTRACT	**5. CONTRACTUAL AUTHORIZATION**	**FILE OPPOSITE SPEC. PAGE NO.** 18
AF04(694)-58	CCN24 (N58-315)*	

6. PRODUCTION EFFECTIVITY:

NA225192 and subsequent.

7. EFFECT OF CHANGE ON SPECIFICATION CONTENT:

Paragraph 3.5.3 Weights (continued)

Accessory equipment	Estimated Weight, lb
Start system	96
Dry weight total	96
Wet weight total	331
Wet weight burnout	128

* NOTE: Previously proposed and authorized as noted.

TABLE OF CONTENTS

Paragraph		Page
1.	SCOPE	1
1.1	Scope	1
1.2	Classification	1
1.2.1	Function	1
2.	APPLICABLE DOCUMENTS	2
3.	REQUIREMENTS	4
3.1	General	4
3.1.1	Model Specification	4
3.1.2	Qualification and Acceptance	4
3.2	Mockup	4
3.2.1	Rocket engine changes	4
3.3	Performance characteristics	4
3.3.1	Rocket engine operating regimes	5
3.3.2	Ratings	5
3.3.3	Estimates	7
3.3.4	Components	7
3.3.5	Starting	7
3.3.6	Shutdown	8
3.3.7	Malfunction	8
3.3.8	External power	9
3.3.9	Propellants and fluids	10
3.3.10	Control	12
3.4	Environmental and load factors	14
3.4.1	Environmental conditions	14
3.4.2	Flight and ground loading conditions	15
3.4.3	Limiting zone temperatures	16

TABLE OF CONTENTS

Paragraph		Page
3.5	Drawings and data	17
3.5.1	Before contract	17
3.5.2	After contract	17
3.5.3	Weights	18
3.5.4	Overall dimensions	18
3.6	Components and systems	18
3.6.1	Propellant and other fluids systems	19
3.6.2	Power control	20
3.6.3	Electric system	22
3.6.4	Ignition system	24
3.6.5	Lubrication system	24
3.6.6	Thrust chamber assembly	25
3.6.7	Tanks	27
3.6.8	Burst diaphragms	28
3.6.9	Accessory drives	28
3.6.10	Accessory equipment	28
3.7	Fabrication	28
3.7.1	Materials	28
3.7.2	Processes	29
3.7.3	Standards	30
3.7.4	Parts list	31
3.7.5	Changes in design	31
3.8	Identification of product	32
3.8.1	Connections	32
3.8.2	Components	32
3.9	General additional information	33
3.9.1	Propellant utilization	33
3.9.2	Thrust alignment	33

TABLE OF CONTENTS

	Paragraph		Page
	4.	QUALITY ASSURANCE PROVISIONS	34
	4.1	Classification of tests	34
	4.2	Tests and test methods	34
	4.2.1	Alternate test fluids	34
	4.2.2	Qualification tests	34
	4.2.3	Preliminary Flight Rating tests	36
(6626)	1.	SCOPE	36
(6626)	2.	APPLICABLE DOCUMENTS	36
(6626)	3.	REQUIREMENTS	36
(6626)	3.1	Reports: rocket engine and components	36
(6626)	3.1.1	General	37
(6626)	3.1.2	Preliminary reports	38
(6626)	3.1.3	Final report	39
(6626)	3.1.4	Number and distribution of copies	44
(6626)	3.2	Disposition of Preliminary Flight Rating Test data	44
(6626)	4.	QUALITY ASSURANCE PROVISIONS	44
(6626)	4.1	General	44
(6626)	4.1.1	Test apparatus and procedures	45
(6626)	4.1.2	Test conditions	46
(6626)	4.1.3	Parts failure and replacement	47
(6626)	4.2	Rocket engine inspections and tests	48
(6626)	4.2.1	Rocket engine tests	48
(6626)	4.2.2	Rocket engine inspection after test	53
(6626)	4.3	Component inspection and tests	53
(6626)	4.3.1	Previous component qualification	53
(6626)	4.3.2	Component inspection before tests	53
(6626)	4.3.3	Component tests	54
(6626)	4.3.4	Component inspection after tests	56

TABLE OF CONTENTS

	Paragraph		Page
(6626)	5.	PREPARATION FOR DELIVERY	57
(6626)	6.	NOTES	57
(6626)	6.1	Intended use	57
(6626)	6.2	Definitions and symbols	57
	4.2.4	Acceptance tests	57
(5152)	1.	SCOPE	57
(5152)	2.	APPLICABLE DOCUMENTS	57
(5152)	3.	REQUIREMENTS	58
(5152)	3.1	Contractor's instructions, specifications and drawings	58
(5152)	3.1.1	Specifications	58
(5152)	3.1.2	Component calibration	58
(5152)	3.1.3	Availability	59
(5152)	3.2	Acceptance test data	59
(5152)	3.2.1	General	59
(5152)	3.2.2	Log	59
(5152)	4.	QUALITY ASSURANCE PROVISIONS	61
(5152)	4.1	General	61
(5152)	4.1.1	Test apparatus and procedures	62
(5152)	4.1.3	Test temperatures	62
(5152)	4.2	Acceptance tests	63
(5152)	4.2.1	Rocket engine inspection before acceptance tests	64
(5152)	4.2.2	Rocket engine tests.- Schedule "A".	64
(5152)	4.2.3	Rocket engine and component tests.- Schedule "B".	66
(5152)	4.2.4	Acceptance conditions	66
(5152)	4.2.5	Rocket engine inspection test	67
(5152)	4.2.6	Rejection and retest	68

TABLE OF CONTENTS

	Paragraph		Page
(5152)	5.	PREPARATION FOR DELIVERY	68
(5152)	6.	NOTES	68
(5152)	6.1	Symbols and definitions	68
	5.	PREPARATION FOR DELIVERY	71
	5.1	Application	71
	5.2	Storage, shipment, and delivery	71
	6.	NOTES	72
	6.1	Intended use	72
	6.2	Symbols and definitions	72
	6.2.1	Definitions	72
	6.2.2	Symbols	79
	6.3	Rocket engine mockup procedure	80
	6.4	Design and installation criteria	80

LIST OF ILLUSTRATIONS

Figures		Page
1a	Estimated Nominal Altitude Performance YLR105-NA-7	81
1b	Estimated Minimum Engine Specific Impulse vs Engine Mixture Ratio YLR105-NA-7	82
2	Estimated Sea Level Thrust & Thrust Coefficient vs Chamber Nozzle Stagnation Pressure at Nominal Mixture Ratio YLR105-NA-7 Thrust Chamber	83
3	Estimated Characteristic Velocity vs Sea Level Thrust at Nominal Mixture Ratio YLR105-NA-7 Thrust Chamber	84
4	Estimated Sea-Level Specific Impulse vs Thrust at Nominal Mixture Ratio YLR105-NA-7 Thrust Chamber	85
5a	Developed Head vs Volumetric Flowrate at Constant Speed YLR105-NA-7 Fuel Pump	86
5b	Developed Head vs Volumetric Flowrate at Constant Speed YLR105-NA-7 Oxidizer Pump	87
6a	Cavitation Characteristics at Constant Speed and Capacity YLR105-NA-7 Oxidizer Pump	88
6b	Cavitation Characteristic at Constant Speed and Capacity YLR105-NA-7 Fuel Pump	89
7	Starting Sequence YLR89-NA-7, When Operated with YLR105-NA-7 and YLR101-NA-15	90
8	Cutoff Sequence YLR105-NA-7, When Operated with YLR89-NA-7 and YLR101-NA-15	91
9	Power Control Arrangement, YLR105-NA-7, When Operated with YLR89-NA-7 and YLR101-NA-15	92
10	Schematic Rocket Engine Flow Diagram, YLR105-NA-7, when Operated with YLR89-NA-7 and YLR101-NA-15	93
11	Start Power Control Interrelation with Rocket Engine, Functional Block Diagram of YLR89-NA-7 When Operated with YLR105-NA-7 and YLR101-NA-15	94

LIST OF ILLUSTRATIONS

Figures		Page
12	Shutdown Power Control Interrelation with Rocket Engine, Functional Block Diagram of YLR89-NA-7 When Operated with YLR105-NA-7 and YLR101-NA-15	95
13	Fuel (RP-1) Density vs Temperature	96
14	Liquid Oxygen Density vs Temperature	97
15	YLR105-NA-7 Thrust Buildup Specification Altitudes: Zero to 10,000 feet	98
16	Limits of Propellant-Utilization Valve Angle vs Engine Mixture Ratio at Sea-Level Standard Conditions	99

TABLES

Tables		Page
I	YLR105-NA-7 Rocket Engine Rating at Standard Sea Level Static Conditions	6
II	Alternate Test Fluids	35

1. SCOPE

1.1 Scope.- This specification covers the requirements for the YLR105-NA-7 liquid rocket engine.

1.2 Classification.- The rocket engine is a calibrated, fixed-thrust, gimbal-mounted bipropellant rocket engine with a nominal sea-level thrust rating of 57,000 pounds. Propellants shall be liquid oxygen and hydrocarbon fuel. The propellants shall be supplied to the thrust chamber from a turbopump powered by a gas generator using the same propellant combination as the thrust chamber. The single thrust chamber shall have an exhaust-nozzle expansion ratio of 25:1, and shall be regeneratively cooled with its fuel as the heat transfer medium.

1.2.1 Function.- The sustainer engine is designed to operate with the YLR89-NA-7 and the YLR101-NA-15 engines not a part of this specification. The interrelation of these rocket engines is as outlined in Figures 7, 8, 9, 10, 11, and 12. The sustainer engine in addition to providing thrust during launch and at altitude provides propellants to the vernier engine, including vernier solo operation, provides pneumatic control for the vernier engine, and provides for the distribution of dc electrical power to the booster engine. In addition the sustainer engine also provides propellants for starting the booster and vernier engine.

2. APPLICABLE DOCUMENTS

2.1 Applicable documents.- The following documents of the exact issue shown form a part of this specification to the extent specified herein.

MILITARY SPECIFICATIONS

MIL-E-5149A 26 March 1956	Engine, Rocket, Liquid Propellant General Specifications for
MIL-H-5606A 21 February 1957	Hydraulic Fluid, Petroleum Base, Aircraft and Ordnance
MIL-L-6086A 30 March 1950	Lubricating Oil, Gear, Petroleum Base
MIL-P-25508D (USAF) 16 March 1962	Propellant Oxygen
MIL-R-25576B 23 Ja uary 1959	Rocket Fuel RP-1
MIL-P-27401A 7 November 1960	Propellant, Nitrogen, Pressurizing
MIL-D-70327 27 March 1962	Engineering Drawings and Associated Lists

AIR FORCE - NAVY AERONAUTICAL BULLETINS

No. 343 j 24 May 1957	Specifications and Standards Applicable to Aircraft Engines and Propellers, Use of
No. 438 a 16 March 1959	Age Controls for Synthetic Rubber Parts

AIR FORCE DOCUMENTS

AFBM Exhibit 58-20A 1 December 19^0	Gas, Fluid and Electrical Conduit Line Identification for Use in Missile and Space Systems

2.1 (Continued)

AF/BSD Exhibit 61-3A 1 September 1961	Specifications for Permissible Contamination Limits and Inspection Criteria for Liquid Oxygen, Liquid Nitrogen, Fuel, Gaseous Oxygen, Gaseous Nitrogen, Instrument Air and Helium, Components, Handling Systems and Fluids Use Limits

ROCKETDYNE DOCUMENT

R-3469 18 July 1962	Associate System Contractor Responsibility for Use With Rocketdyne Propulsion System Contracts

SPACE TECHNOLOGY LABORATORIES

GM 6300.8-565B 2 February 1962	Contamination Limits and Evaluation Methods for Hydraulic Systems and Components, WS-107A-1, Weapons System.

3. REQUIREMENTS

3.1 General.-

3.1.1 Model Specification.- The model specification has been prepared using MIL-E-5150 as a guide with applicable provisions of MIL-E-5149 incorporated. Publications referenced in the model specification and contained in ANA Bulletin 343 shall be applicable as provided by the bulletin. Where the requirements of this model specification and those of the documents listed in Section 2 are at variance, the requirements of the model specification shall govern.

3.1.2 Qualification and acceptance.- The qualification and acceptance of any engine shall be in accordance with the tests specified in Section 4 of this specification.

3.2 Mockup - The contractor shall prepare a full-scale mockup of the rocket engine when required by contract.

3.2.1 Rocket engine changes.- The Using Service shall be notified of changes to the rocket engine features affecting the installation made after approval of a mockup or the drawing package specified in paragraph 3.5. Any mockup specifically required by contract shall be kept current with approved changes for the duration of the production contract, unless otherwise authorized. Changes required by the procuring activity shall be subject to negotiations.

3.3 Performance characteristics.- The ratings, data, and curves shown are based on standard-sea-level static conditions, unless otherwise noted.

3.3.1 Rocket engine operating regimes.-

3.3.1.1 Altitudes and temperatures.- The rocket engine shall start, operate and stop throughout the design range specified herein under the following conditions:

3.3.1.1.1 Static exposure.- The dry rocket engine shall not suffer any detrimental effects when exposed, in a nonoperating condition, to a temperature range of minus 65 to plus 160 F.

3.3.1.1.2 Operation.- The rocket engine shall operate for the rated duration and stop satisfactorily at any altitude, and shall start at any altitude up to 10,000 feet provided that it is within the ambient temperature range of minus 30 to plus 130 F at start and that the fluids are supplied within the temperature ranges specified in paragraphs 3.3.8, 3.3.9, and 3.6.1.

3.3.1.2 Attitudes.- The rocket engine shall start only in a vertical position. The rocket engine shall operate and stop satisfactorily throughout any flight path in which the acceleration vector of the thrust chambers does not depart by more than the effective gimbal angle from the longitudinal centerline of the vehicle, and when operated within the load conditions specified in paragraph 3.4.2.3.

3.3.2 Ratings.- The performance ratings shall be as listed in Table I. The data shall be based on the use of fuel at the specification density midpoint, 50.45 lb/ft^3, Figure 13, and oxidizer at the ambient sea level density, 71.38 lbs/ft^3, Figure 14.

TABLE I

YLR105-NA-7 ROCKET ENGINE RATING AT STANDARD SEA LEVEL STATIC CONDITIONS

Engine thrust lbs (a)	Engine instantaneous specific impulse, seconds, minimum (b)	Engine altitude (0 psia ambient) specific impulse, seconds, calculated min. (b)	Engine altitude (0 psia ambient) thrust, pounds, calculated minimum	Nominal chamber pressure psia	Engine mixture ratio O/F (c)	Engine inlet pressures, psia		Effective duration seconds
						Fuel	Oxidizer	
58,710 Max. 57,000 55,290 Min.	213	305.7	78,280	662	2.61 Max. 2.27 1.93 Min.	77	53	300

(a) Engine thrust considered parallel to thrust chamber axis. The engine thrust values are corrected to rated engine inlet conditions.

(b) Specific impulse includes gas generator flow rates. YLR105-NA-7 specific impulse excludes the YLR101-NA-15 flow rate. The mean specific impulse of an individual engine, corrected to nominal rated thrust, nominal rated mixture ratio, and rated engine inlet conditions, shall be equal to or greater than the minimum specified.

(c) Mixture ratio values are corrected to rated engine inlet conditions, and nominal rated thrust at nominal rated mixture ratio, and includes flow rate to the gas generator and to the YLR101-NA-15 engine and engine oxidizer tank refill. Mixture ratio range represents maximum engine capability.

YLR101-NA-15 Nominal Pump-Fed Bleed Pressure, psia

Fuel	Oxidizer
845	769

3.3.3 Estimates.- The estimated nominal altitude performance versus propellant flow rate, specific impulse, and thrust is shown in Figure 1, based on the inlet pressures of Table I, and the propellant densities of paragraph 3.3.2.

3.3.4 Components.- Curves shown in Figures 2 to 6 inclusive shall constitute a part of this specification.

3.3.4.1 Thrust chamber.-

(a) Figure 2, Estimated sea-level thrust and thrust coefficient vs chamber nozzle stagnation pressure at nominal mixture ratio.
(b) Figure 3, Estimated characteristic velocity vs sea-level thrust at nominal mixture ratio.
(c) Figure 4, Estimated sea-level specific impulse vs thrust at nominal mixture ratio.

3.3.4.2 Pumps.-

(a) Figures 5a and 5b, Developed head vs volumetric flow rate at constant speed.
(b) Figures 6a and 6b, Cavitation characteristics at constant speed and capacity.

3.3.5 Starting.- The rocket engine is capable of being ground started. Ground Support Equipment, in conjunction with the YLR89-NA-7 and the YLR101-NA-15 engines or their simulated equivalents, not a part of this specification, must be utilized to effect ground starting. The rocket engine provides the propellants for starting the YLR89-NA-7 and the YLR101-NA-15. The start system provides for one start only without reservicing. The start sequence is as set forth in Figure 7.

3.3.6 Shutdown.- Provisions incorporated for cutoff shall ensure that a positive and safe shutdown can be reliably achieved under all normal operating conditions. The rocket engine incorporates provisions for signaling cutoff for the YLR89-NA-7 and the YLR101-NA-15 engines. The shutdown sequence is as set forth in Figure 8.

3.3.6.1 Normal shutdown.- The cutoff sequence is initiated by a signal to the command cutoff connection at the Engine Relay Box. Detailed sequencing as shown in 3.6.3.5.

3.3.7 Malfunction.- Supplied with specification propellants the rocket engine shall, under any single condition of malfunction, start and operate in a stable, safe, and reliable manner, or shutdown without presenting a hazardous condition that could cause damage to the vehicle, except that the malfunction shutdown may be restricted or eliminated after set performance has been attained, and except that the malfunction shutdown sensing circuit shall be eliminated after the missile is irretrievably committed for flight. Malfunctions include but are not limited to events such as: power control malfunction, electrical system failure, external power interruption or fluctuation, or fortuitous subjection to conditions exceeding specified operating parameters. Subject to the approval of the procuring activity, where it is found that certain malfunction conditions exist that cannot be overcome without compromising overall operation, malfunctioning shall be reduced to a minimum by designing and developing into each control element the best attainable reliability or by utilizing appropriate checkout procedures. When required by contract, an analysis of pertinent malfunction conditions anticipated in service usage as agreed upon by the procuring activity and the contractor shall be prepared as a separate report.

3.3.7 (Continued)

This analysis shall show that the rocket engine design has fulfilled the safety requirements as specified in this paragraph and shall be submitted to the procuring activity prior to performance of "Rocket engine inspection and test" of Specification MIL-E-5151 and MIL-E-6626.

3.3.8 External power.- External power shall be available to the rocket engine as follows:

3.3.8.1 Pneumatic requirements.-

(a) High pressure pneumatic requirements.- 3000 to 1000 psig helium at a temperature range of minus 65 to plus 160 F shall be supplied to the engine pneumatic inlet at the following requirements. All values are maximums.

	Quantity	Flow rate
(1) Standby	---	0.083 lbs/min
(2) Starting	0.34 lbs	0.232 lbs/min
(3) Flight operation	0.393 lbs	0.00132 lbs/sec
(4) YLR101-NA-15 Solo flight	2.5 lbs	0.[illegible]0 lbs/sec

(b) Low pressure pneumatic requirements.- Helium in the pressure range of 40 to 62 psig and at a temperature range of minus 65 to 160 F shall be supplied to the engine lubricant tank pneumatic inlet during flight operation at a maximum rate of 0.00012 lbs/sec. A maximum quantity of 0.036 pounds will be consumed during a rated duration flight.

3.3.8.2 Electrical requirements.- Electrical power in accordance with Specification MIL-E-7894 at the following input requirements. Components and signals.- 25 to 30 volts dc maintained during engine operation for components. The command cutoff signals shall have a minimum current capacity of 0.30 amperes and shall be sustained for a minimum period of 0.10 seconds. Maximum dc power required (watts) is as follows:

	Ground power	Missile power
Start	500	300
Flight	---	250

AC Components.- 120/208 volts ac, 3 phase, 4 wire, 60 to 400 cycles, for heaters during preflight (ground supply). Maximum 1500 watts is required.

3.3.8.3 Hydraulic power.- Hydraulic power shall be supplied to the YLR105-NA-7 engine from the missile within a range of 2500 to 3000 psia at a temperature of plus 50 F to plus 275 F. Maximum transient requirements are 26.5 gpm for the YLR105-NA-7 head suppression and propellant utilization systems. The total volume of the YLR105-NA-7 hydraulic system shall not exceed 90 cubic inches including the accumulator.

3.3.9 Propellants and fluids.-

3.3.9.1 Propellants.-

(a) Oxidizer.- The oxidizer supplied to the engine shall be liquid oxygen in accordance with MIL-P-25508 and shall meet the purity requirements of AFBSD Exhibit 61-3A as implemented by Rocketdyne Report R-3469, Section I.

3.3.9.1 (Continued)

(b) Fuel.- The fuel supplied to the engine shall be rocket engine fuel Grade RP-1 conforming to MIL-R-25576 and shall meet the purity requirements of AFBSD Exhibit 61-3A as implemented by Rocketdyne Report R-3469, Section I.

3.3.9.2 Pressurizing gas.- Helium, Bureau of Mines Grade A, at 3000 to 1000 psig, during engine operation and 3000 to 800 psig during YLR101-NA-15 solo operation; and 40 to 62 psig at a temperature range of minus 65 to plus 160 F shall be supplied to the engine and shall meet the purity requirements of AFBSD Exhibit 61-3A as implemented by Rocketdyne Report R-3469, Section I.

3.3.9.3 Lubricants.- Turbopump gear box lubricant: Oil in accordance with Specification MIL-L-6086A, dated 30 March 1950 and as amended 3 January 1955, supplied within the temperature range of plus 45 F to plus 130 F, at a nominal rate of 1.2 gpm.

3.3.9.4 Other fluids.-

3.3.9.4.1 Pyrophoric fluid.- Pyrophoric fluid shall be supplied in a container to establish thrust chamber ignition.

3.3.9.4.2 Hydraulic fluid.- Hydraulic fluid shall be supplied to the engine from the vehicle within the range of 2500 to 3000 psia at a temperature range of plus 50 to plus 275 F. Maximum transient requirements are 26.5 gpm for the YLR105-NA-7 head suppression and propellant utilization systems. Hydraulic fluid shall be in accordance with MIL-U-5606 and shall meet the purity requirements of Space Technology Laboratories' document GM 6300 .8-565B as implemented by the Rocketdyne Report R-3469, Section V. The total volume of the MA-5 hydraulic system shall not exceed 90 cubic inches including the accumulator.

3.3.9.5 Leakage.- External or internal leakage of the fluids shall not be permitted where such leakage will impair proper functioning of or cause damage to the rocket engine or the missile and its components or will endanger personnel. Leakage limits shall be as shown on the applicable drawings of paragraph 3.5.

3.3.10 Control.-

3.3.10.1 Accuracy.- The control shall be such that the rocket engine shall operate within the limits specified in paragraph 3.3.2.

3.3.10.1.1 Mixture ratio.- The mixture ratio shall be controlled within safe operating limits during mainstage operation and during thrust increase and decrease. The YLR101-NA-15 mixture ratio as affected by engine mixture ratio change shall be controlled within safe operating limits during mainstage operation. During flight operation the mixture ratio specified in Table I shall be capable of varying plus or minus 15 percent to comply with the requirements of the propellant utilization system of the missile. Means of limiting the mixture ratio to safe engine operating values shall be incorporated in the propellant-utilization system, not a part of this specification. The PU-valve angle versus mixture ratio curve shall have the slope characteristics noted below (see Figure 16):

(1) The ratio of the slopes at mixture-ratio values of 1.93 and 2.61 shall not exceed 10:1 when the slope ratio is determined from tangents drawn to the curve at the specified mixture-ratio values.

(2) The slope at a mixture ratio value of 2.27 shall be no less than 15.7 and no greater than 36.7 degrees valve angle per mixture ratio unit.

3.3.10.1.1 (Continued)

(3) The PU-valve-angle travel shall be greater than 5 degrees and less than 23 degrees between mixture-ratio values of 2.27 and 2.61, and greater than 10 degrees and less than 27 degrees between mixture-ratio values of 2.27 and 1.93.

3.3.10.2 <u>Thrust</u>.- There are no intermediate controlled-thrust settings except for those resulting from propellant utilization mixture ratio variation.

3.3.10.2.1 <u>Change rate</u>.-

3.3.10.2.2 <u>Increase</u>.- The time interval between "engine tanks pressurized" and "auxiliary system complete" shall not exceed 30 seconds. The time interval between "auxiliary system complete" and "mainstage control" shall not exceed 4.0 seconds. The time interval between "mainstage control" and 90 percent of rated thrust shall not exceed 2.0 seconds. Thrust buildup shall be in accordance with Figures 7 and 15.

3.3.10.2.3 <u>Decrease</u>.- Thrust decrease values are site values within the range of 0 to 5000 feet altitude.

3.3.10.2.3.1 <u>Cutoff</u>.- The control system shall provide the following engine cutoff impulse at site conditions as determined from each performance test. The mixture ratio at the time of cutoff signal shall be 2.27 plus or minus 5 percent.

	Delay Circuit Inactive	Delay Circuit Active
Minimum	4000 lb/sec	4450 lb/sec
Maximum	8200 lb/sec	9650 lb/sec

3.3.10.3 Stability.- Thrust oscillation during transient conditions shall not produce thrust peaks of greater than 110 percent of rated thrust at frequencies below 150 cycles per second. Superimposed on this fundamental frequency may be high-frequency oscillation above 150 cycles per second and not greater in magnitude than plus or minus 3 percent of rated thrust.

3.3.10.4 Starting.-

3.3.10.4.1 Procedure.- The starting procedure for the rocket engine shall be as specified in paragraph 3.3.5.

3.3.10.4.2 Time.- The elapsed time from mainstage control to 90 percent of maximum rated thrust shall be as specified in paragraph 3.3.10.2.2.

3.3.10.5 Shutdown.-

3.3.10.5.1 Procedure.- The shutdown procedure shall be as specified in paragraph 3.3.6.

3.3.10.5.2 Time.- The thrust decay shall be as specified in paragraph 3.3.10.2.3.

3.3.10.6 Auxiliary functions.- Not applicable.

3.4 Environmental and load factors.-

3.4.1 Environmental conditions.- The rocket engine shall not suffer any detrimental effects during and after any condition of environment that has been demonstrated by the environmental qualification tests of paragraph 4.2.2.

3.4.1.1 Temperature range.- The rocket engine under field storage conditions, in the protective container, shall not suffer any detrimental effects when exposed to the temperature range of minus 65 to plus 160 F.

3.4.1.1.1 Engine relay box.- Components within the engine relay box shall perform in flight as described in paragraph 3.3.6 when exposed to temperatures in the range of plus 40 to plus 80 F.

3.4.1.2 Vibration.- The rocket engine shall withstand all vibration encountered in normal usage without deleterious effect on the rocket engine or impairment of its serviceability.

3.4.1.2.1 Engine relay box.- Components within the YLR105-NA-7 engine relay box shall perform in flight as described in paragraph 3.3.6 when exposed to vibration conditions not to exceed 8 g's, or plus or minus 1/4-inch amplitude in the frequency range between 20 and 2000 cps, in each of three mutually perpendicular planes.

3.4.2 Flight and ground loading conditions.- The rocket engine and its supports shall withstand, without permanent deformation or failure, the largest forces resulting from all critical combinations of load factors specified in paragraph 3.6.6. For design purposes, the ultimate strength shall provide for a minimum of 1.5 times the forces resulting from the loading conditions. The rocket engine shall be designed to withstand 4.0 g handling loads applied in any direction. Demonstration of these requirements may be waived at the discretion of the procuring activity when acceptable substantiating analytical data are furnished by the contractor.

3.4.2.1 Aircraft rocket engines.- Not applicable.

3.4.2.2 Aircraft launched missile rocket engines.- Not applicable.

3.4.2.3 Vehicle rocket engine.- The rocket engine and its supports while meeting the gimbaling requirements of paragraph 3.6.6, shall operate satisfactorily, without permanent deformation or failure, under any one of the following load conditions:

(a) 12.0 g's parallel to the direction of flight and 1.25 g's perpendicular to the direction of flight.

(b) 10.0 g's parallel to the direction of flight and 1.5 g's perpendicular to the direction of flight.

(c) 2.5 g's parallel to the direction of flight and 3.0 g's perpendicular to the direction of flight.

3.4.3 Limiting zone temperatures.- No heating or cooling provisions are required from the vehicle during flight. Requirements for heaters during preflight shall be in accordance with paragraph 3.3.8.

3.5 Drawings and data.- The following Rocketdyne drawings and data shall form a part of this specification:

(a)	Engine assembly (sustainer)	1002001
(b)	Markings installation	1002004
(c)	Control system installation (gas generator system)	3002001
(d)	Propellant feed system installation	4002001
(e)	Lubrication system installation	5502010
(f)	Electrical system installation	5002001
(g)	Engine loose equipment	6502001
(h)	Expendable loose equipment	6502002
(i)	Pyrotechnic loose equipment	6502003
(j)	Hydraulic system installation	6002001
(k)	Exhaust system installation	3002010

Model identification numbers

ENGINE AND MPL	MD NUMBERS
YLR105-NA-7	7x11x13 15x17x20 23x25 28x31

3.5.1 Before contract.- Not applicable.

3.5.2 After contract.- Not applicable.

3.5.3 **Weights.**- The dry weight of the YLR105-NA-7 shall not exceed 1027 pounds. The wet weight after normal shutdown shall not exceed 1151 pounds. The wet weight with all systems filled to capacity shall not exceed 1416 pounds.

Assembly	Estimated Weight, lb
Thrust chamber	367
Mount, gimbal assembly	18
Turbopump installation	229
Oxidizer system	46
Fuel system	40
Gas generator system	45
Lubrication system	21
Electrical system	22
Pneumatic system	7
Exhaust system	85
Hydraulic system	45
Ignition system	6
Dry weight total	931
Wet weight total	1085
Wet weight burnout	1023
Accessory equipment	
Start system	96
Dry weight total	96
Wet weight total	331
Wet weight burnout	128

3.5.4 **Overall dimensions.**- The overall dimensions of the rocket engine shall be as shown on the applicable drawings of paragraph 3.5.

3.6 **Components and systems.**-

3.6.1 Propellant and other fluids systems.- The rocket engine shall function satisfactorily when the propellants and other fluids are supplied within the following start conditions: Fuel at plus 32 to 80 F and liquid oxygen at a maximum of 10 F above the ambient sea level boiling point. Engine start and operation are subject to the conditions set forth in paragraph 3.6.1.1. Other fluids must be in accordance with paragraps 3.3.8 and 3.3.9.

3.6.1.1 Pump and drive system.- The turbopump shall have a pressure compounded, axial-flow, two stage turbine driving two single entry centrifugal propellant pumps. The turbines shall be hot gas driven. In addition to supplying propellants to the thrust chamber and gas generator, the turbopump shall supply propellants for the YLR101-NA-15 engine and shall supply oxidizer for replenishment of the engine oxidizer tank. The turbopump shall operate satisfactorily with the following inlet conditions. The minimum net positive suction head (NPSH) referenced to the pump inlet centerline shall be as follows:

	Liquid Oxygen, feet	Fuel, feet
(a) Starting	70	150
(b) Rated thrust operation	20	85

The rocket engine shall be capable of safe operation and recovery, with a resultant loss in performance, if the oxidizer NPSH is lowered to 18 feet for a maximum period of 5 seconds during staging.

3.6.1.1.1 Turbine-exhaust connection.- The turbine exhaust system shall be as shown on the exhaust systems installation drawings.

3.6.1.2 Fluid drainage.- Drainage of like fluid system elements shall be collected at common points. All drainage connections shall be indicated on the installation drawings. The normal drainage procedure shall consist of fluid runoff, flushing with solvent where applicable, and drying with an inert gas. At the completion of this procedure the amount of fluids remaining shall be negligible.

3.6.1.3 Lines and fittings.- The minimum and maximum torque values shall be as specified in Drawing AND10064 for the sizes or types of lines and fittings except as specified on the contractor's drawings.

3.6.1.4 Filters.- Filters shall be as shown on applicable drawings of paragraph 3.5 and shall be of a design and fabrication that will not permit passage of solid impurities which will affect engine operation when fluids are supplied within the limits of paragraph 3.3.9.

3.6.1.5 Filler connections.- Not applicable.

3.6.2 Power control.- The controls shall provide for starting, operating, and stopping the rocket engine in accordance with the requirements of this specification. During full-thrust operation at standard sea-level static conditions, the design requirements of the controls shall be to maintain performance as shown in Table I.

3.6.2.1 Preflight check.- Preflight check shall be obtained through the ground support equipment, not a part of this specification. All control circuits may then be checked by direct or simulated operation.

3.6.2.1.1 External test connections.- Noninterchangeable test connections as required for safety shall be provided for ground checking of significant sequencing and emergency devices. Details of the connections shall be presented on the installation drawings.

3.6.2.2 Indication.- Switches shall be provided on the rocket engine to give an indication of open and closed positions of the main propellant valves, and gas generator valves. A pressure switch shall be provided to give an indication of effective mainstage thrust. An indication signal of the propellant utilization valve angle position from a given reference point shall be provided. Circuits shall be provided on the rocket engine to give an indication of the dc voltage at the relay box busses, and the condition of control circuits as necessary. Provisions shall be made on the rocket engine to give an indication of cutoff. Indicating circuits may be disconnected at lift-off.

3.6.2.3 Calibration.- All field replaceable controls shall have capability of being installed and used without field calibration.

3.6.2.4 Interrelation with rocket engine.-

(a) The "Power control arrangement" is as shown in Figure 9.
(b) The "Sequence of rocket engine operation" is as shown in Figures 11 and 12.

3.6.2.4.1 Performance selector (For aircraft).- Not applicable.

3.6.2.5 Starting.- The starting sequence shall be automatic upon initiation by ground support equipment, not a part of this specification.

3.6.2.5.1 Fixed thrust rocket engines.- Not applicable.

3.6.2.5.2 Variable thrust rocket engines.- Not applicable.

3.6.2.6 Control adjustment.- The rocket engine does not provide for variable thrust operation.

3.6.3 Electric system.-

3.6.3.1 Electrical power.- All components using electrical power from the vehicle power system shall be consistent with the requirements of paragraph 3.3.8.2 herein.

3.6.3.2 Radio interference.- Electrical components shall, during flight operation, not cause radio interference beyond the limits specified in Specification MIL-I-6181B except for three(3) milliseconds maximum during shutdown of the YLR105-NA-7 and the YLR101-NA-15.

3.6.3.3 Ignition proof.- Electrical components shall not ignite any explosive mixture surrounding the equipment.

3.6.3.4 Connectors and cable.- It shall be possible to connect or disconnect electrical connectors and to flex electrical conductors as necessary for routine maintenance without damage, at a temperature of minus 65 F. All electrical wiring shall be capable of withstanding 400 F for the [illegible] duration without causing malfunction.

3.6.3.5 Engine relay box.- The engine relay box shall consist of the necessary control circuits to provide electrical control for engine maintenance operation and cutoff upon receipt of a cutoff signal to the command cutoff connection. In addition the engine relay box provides upon receipt of a signal to the command cutoff connections cutoff signals for the YLR89-NA-7 and the YLR101-NA-15. A circuit for controlling the repressurization of the engine tanks upon command is provided. Cutoff delay circuits (desensitizing network) are provided for all cutoff signals. The YLR89-NA-7 cutoff delay network shall delay engine shutdown by 55 plus or minus 30 milliseconds. In addition, the cutoff circuit shall be insensitive to sustained signals of less than 25 milliseconds duration. The YLR105-NA-7 cutoff delay network shall delay engine shutdown by 22 plus or minus 8 milliseconds. In addition, the cutoff circuit shall be insensitive to sustained signals of less than 14 milliseconds duration. The YLR101-NA-15 cutoff delay network shall delay engine shutdown by 55 plus or minus 18 milliseconds. In addition, the cutoff circuit shall be insensitive to sustained signals of less than 37 milliseconds duration.

NOTE: YLR89-NA-7 cutoff initiated prior to YLR105-NA-7 or YLR101-NA-15 cutoff will effect YLR89-NA-7 cutoff. YLR105-NA-7 cutoff initiated prior to YLR89-NA-7 or YLR101-NA-15 cutoff will effect YLR89-NA-7 and YLR105-NA-7 cutoff simultaneously. YLR101-NA-15 cutoff initiated prior to YLR89-NA-7 or YLR105-NA-7 cutoff will effect YLR89-NA-7, YLR105-NA-7, and YLR101-NA-15 cutoff simultaneously.

3.6.4 Ignition system.-

(a) Main thrust chamber.- The main thrust chamber ignition system shall consist of a pyrophoric, fluid-filled, pressure-actuated container used in a hypergolic application. The thrust chamber shall be provided with an igniter fuel source separate from the main propellant source.

(b) Gas generator.- The gas generator ignition system shall consist of an electrically-fired pyrotechnic and electric circuitry capable of firing the igniter and sensing the firing thereof. A combustible mixture of propellants shall be provided to the gas generator combustion chambers upon satisfactory igniter burning. The sequencing of gas generator igniter firing and propellant admission shall be controlled by the ground support equipment electrical control system.

3.6.4.1 High-tension lead assembly.- Not applicable.

3.6.5 Lubrication system.- The lubrication tank, as specified in paragraph 3.6.7, shall be pressurized by pneumatic pressure referenced in 3.3.8.1 The tank shall supply oil to the rocket engine positive displacement type oil pump discharging at 500 to 1000 psig to oil jet lines. Oil jets are directed at the bearings and the disengaging meshes of the turbopump gears and then the oil is ducted overboard. The lubrication pump is an integral part of the turbopump.

3.6.6 Thrust chamber assembly.- The thrust chamber shall be gimbaled to permit thrust chamber movement as follows. Gimbaling actuators shall not be furnished with the engines.

(a) The thrust chamber shall be gimbal-mounted at the approximate center of the injector dome. The turbopump, propellant valves, and turbine exhaust are mounted to the thrust chamber so that the rocket engine is essentially gimbal mounted. The gimbal mount shall permit thrust chamber movement as follows:

(1) Angle of displacement of the effective thrust vector from the normal to the plane of the gimbal axis shall be 4.5 degrees.

(2) Total restraining moment from gimbal point friction alone, 1000 ft-lb maximum at sea-level.

(3) The engine shall be capable of withstanding a lateral acceleration of 40 ft/sec^2.

(4) The moment of inertia and any combination of all restraining torques including hose restraint, off-center cg location, gimbal friction, specified thrust misalignment, and lateral acceleration, and longitudinal acceleration, shall be such that with a force of 14,000 pounds on actuator number 1 and 9500 pounds on actuator number 2, on arms of 10 inches the thrust chamber shall accelerate a minimum of 110 deg/sec^2 at sea level. The hose restraint for the fuel and LOX lines shall not exceed 3000 in-pounds maximum. The maximum acceleration shall be 4900 degrees/sec^2.

(5) The thrust chamber shall be capable of withstanding a maximum force of 19,000 pounds on both actuators on arms of 10 inches from the gimbal axis.

3.6.6 (Continued)

(b) The physical characteristics of the wet gimbaled mass shall be as follows:

(1) The nominal value for the moment of inertia of the wet gimbaled mass is 429 slug-ft^2 about the gimbal point yaw axis. The variation of this moment of inertia from engine to engine shall not exceed plus or minus 7 percent of the nominal value.

(2) The nominal wet weight of the gimbaled mass is 1046 pounds about the pitch and yax axis. The variation of this mass from engine to engine shall not exceed plus or minus 5 percent of the nominal value.

(3) The nominal longitudinal wet cg location of the gimbaled mass with respect to the gimbal point is 31.6 inches aft. This distance shall not vary from engine to engine more than plus or minus 5 percent of the nominal value.

(4) The lateral location of the cg of the wet gimbaled mass relative to the gimbal pitch axis is approximately 4.4 inches.

(5) The vertical location of the cg of the wet gimbaled mass relative to the gimbal yaw axis is approximately minus 1.0 inch.

(c) Loads acting simultaneously or otherwise on the turbopump oxidizer and fuel-inlet flanges, produced by missile propellant ducting, shall not exceed (for each inlet flange):

3.6.6 (Continued)

(1) 3000 inch-pounds torque acting about axis of inlet-flange opening, and 3000 inch-pounds torque acting about each of any two axes normal to each other and to axis of flange.

(2) Shear load of 200 pounds acting parallel to each of any two axes normal to each other and to axis of inlet flange.

(3) Load of 1000 pounds acting parallel to axis of inlet-flange opening.

3.6.6.1 Propellant accumulation.- Propellants remaining in the rocket engine aft of the turbopump inlet flanges after shutdown are as specified in paragraph 3.5.3.

3.6.7 Tanks.- Tanks shall be designed, manufactured, and tested to standards consistent with the load requirements of paragraph 3.4.2, and the operating pressures indicated below. The usable fluid capacity of the fuel tank is dependent on a missile system ullage fitting, not a part of this specification. Specification MIL-T-5208A is not applicable.

	Usable Fluid Capacity (approximately) cu. in.	Operating Pressures psig (nominal)
Lubrication tank	1550	57
Oxidizer tank	4140	600
Fuel tank	2920	600

3.6.7 (Continued)

The engine oxidizer tank shall be replenished from the turbopump through the missile interconnect system, not a part of this specification, during the first 120 seconds of operation provided the refill pressure at the inlet to the oxidizer tank fill and check valve is a minimum of 640 psig. The engine fuel tank shall be replenished from the YLR89-NA-7 during the first 120 seconds of operation. The engine propellant tanks, after replenishment, shall be repressurized a minimum of 5 seconds prior to YLR105-NA-7 command cutoff.

3.6.8 Burst diaphragms.- Burst diaphragms are provided in the thrust chamber igniter cartridge.

3.6.9 Accessory drives.- Accessory drives shall be provided as follows: One accessory pad shall be provided on the turbopump for attachment of missile accessories. The pad shall conform to AND20001-XI-C with the following exceptions:

(a) Direction of rotation, clockwise facing engine pad.
(b) No provision for lubrication of accessories.

3.6.10 Accessory equipment.- Not applicable.

3.7 Fabrication.-

3.7.1 Materials.-

3.7.1.1 Quality.- Materials used in the manufacture of the rocket engine shall be of high quality, suitable for the purpose, and shall conform to applicable specifications in accordance with ANA Bulletin No. 343. When contractor's specifications are used for materials which may affect performance or durability of the rocket engine, such specifications will be released to the Government prior to the Qualification tests. The use of non-Governmental specifications shall not constitute waiver of Government inspection.

3.7.1.2 Critical materials.- The use of critical materials shall be held to a minimum. The list of critical materials noted in paragraph entitled "critical materials" of Specification MIL-E-5149A, and the estimated weights thereof based on the finished parts are as follows:

Material	Weight, pounds
(a) Chromium	8
(b) Cobalt	1
(c) Columbium	0
(d) Molybdenum	1
(e) Natural rubber	0
(f) Nickel	50
(g) Tungsten	0

3.7.2 Processes.-

3.7.2.1 Quality.- When contractor's specifications are used for processes which may affect performance or durability of the rocket engine, such specifications will be released to the Government prior to the Qualification tests. The use of non-Governmental specifications shall not constitute waiver of Government inspection.

3.7.2.2 Workmanship.- The workmanship and finish shall be of sufficiently high grade to ensure satisfactory operation, reliability, and durability consistent with the service life and application of the rocket engine.

3.7.2.3 Interchangeability.- All parts having the same manufacturer's part number shall be directly and completely interchangeable with respect to installation and performance except that matched parts or selective fits will be permitted where required. Changes in manufacturer's part numbers shall be governed by the drawing number requirements of Specification MIL-D-70327.

3.7.2.4 Protective treatment.- With the exception of working surfaces and drive pad faces, all parts shall be corrosion resistant or suitably protected.

3.7.3 Standards.-

3.7.3.1 Parts.- AN, JAN, or MIL Standard parts shall be used wherever they are suitable for the purpose, and shall be identified by their Standard part numbers. The use of nonstandard parts will be acceptable only when standard parts have been determined to be unsuitable.

3.7.3.2 Design.- MS and AND Design Standards shall be used wherever applicable.

3.7.3.3 Threads.- Conventional straight screw threads shall conform to the requirements of Specification MIL-S-7742. Tapered pipe threads may be employed only for permanently installed fittings or plugs.

3.7.4 Parts list.- The parts list for the rocket engine which successfully completes the Preliminary Flight Rating tests shall constitute the approved parts list for subsequent engines of the same model. Changes to the approved rocket engine parts list shall be governed by the requirements specified in paragraph 3.7.5.

3.7.5 Changes in design.- Changes, made in the design or materials of parts listed in an approved rocket engine parts list shall be approved in accordance with the provisions of ANA Bulletin No. 391a, as incorporated in the contract.

3.7.5.1 Class I changes.- Definitions shall be as provided in paragraph 3.7.5.

3.7.5.2 Class II changes.- Definitions shall be as provided in paragraph 3.7.5.

3.7.5.3 Approval of changes.- Approval of changes does not relieve the contractor of full responsibility for the results of such changes on rocket engine characteristics.

3.8 Identification of product.- Equipment, assemblies, and parts shall be marked for identification in accordance with MIL-STD-130. The identification data applied to the rocket engine data plate shall be as follows:

Engine, Rocket, Liquid Propellant
Government Model Designation ___*___
Model Specification No. ___*___
Serial No. ___*___
Manufacturer's Part No. ___*___
Contract or Order No. ___*___
Manufacturer's Name or Trade-Mark ___*___
US

* Applicable data to be entered by the contractor.

3.8.1 Connections.- The rocket engine shall be permanently marked to indicate all connections shown on the installation drawing for instrumentation, propellant, and other fluid connections. All fluid lines shall be marked in accordance with AFBM Exhibit 58-20, as implemented by Rocketdyne Report R-3469 Section VI.

3.8.2 Components.- Components shall be clearly marked as follows:

(Nomenclature)
Serial No. ___*___
Stock No. ___*___
Manufacturer's Part No. ___*___
Manufacturer's Name or Trade-Mark ___*___

* Applicable data to be entered by the contractor.

3.8.2.1 Synthetic rubber parts.- Components used in hydrocarbon fluid systems containing synthetic rubber parts shall be marked in accordance with ANA Bulletin No. 438.

3.9 General additional information.-

3.9.1 Propellant utilization.- A means shall be provided to vary the propellant mixture ratio of the rocket engine by plus or minus 15 percent in accordance with a signal from a suitable sensing system, not a part of this specification. Mixture ratio versus specific impulse is as set forth in Figure 1b.

3.9.2 Thrust alignment.-

3.9.2.1 Thrust vector lateral displacement.- The resultant thrust vector of the rocket engine shall pass within 1/8 inch of the gimbal point.

3.9.2.2 Thrust vector angular alignment.- The angular inclination of the resultant rocket engine thrust chambers thrust vector, with respect to the geometric centerline of the thrust chamber, shall not exceed 0.5 degree.

4. QUALITY ASSURANCE PROVISIONS

4.1 Classification of tests.- The testing of liquid propellant rocket engines shall be classified as follows:

(a) Qualification tests.- The Qualification tests are conducted to demonstrate the suitability of an engine model for production.

(b) Preliminary Flight Rating tests.- The Preliminary Flight Rating tests are conducted to demonstrate the suitability of an engine model for use in experimental aircraft or missile flight testing.

(c) Acceptance tests.- The acceptance tests are conducted on engines submitted for acceptance under contract.

(1) Miscellaneous inspection tests.- Various inspection tests and procedures are conducted during the course of manufacture to ensure that adequate quality control is maintained for materials and manufacturing purposes.

4.2 Tests and test methods.-

4.2.1 Alternate test fluids.- Cold calibration, in accordance with Table II.

4.2.2 Qualification tests.- Qualification test requirements shall be as specified by Specification MIL-E-5151, as modified by mutual agreement of the procuring activity and the contractor. Demonstration of these requirements shall be as required by contract.

TABLE II

ALTERNATE TEST FLUIDS

Specified propellant or fluid	Alternate test fluid
Liquid oxygen	Liquid nitrogen
RP-1	Water
Helium	Dry nitrogen in accordance with Specification MIL-P-27401 Grade A, Type I

4.2.3 **Preliminary Flight Rating tests**.- Establishment of a Preliminary Flight Rating for the rocket engine is predicated on prior satisfactory completion of the tests on the MA-2 propulsion system in accordance with Specification MIL-E-6626, except as modified or reiterated hereafter with paragraph numbers identified by (6626).

1. (6626) SCOPE

1.1 (6626) This specification establishes Preliminary Flight Rating test requirements for approving the use of a liquid-propellant rocket engine in a vehicle under restricted usage conditions.

2. (6626) APPLICABLE DOCUMENTS

2.1 (6626) The applicable publications listed in the following bulletin, of the issue specified in the manufacturer's engine model specification, form a part of this specification:

PUBLICATIONS

Air Force - Navy Aeronautical Bulletin

No. 343 Specifications and Standards Applicable to Aircraft Engines and Propellers, Use of

3. (6626) REQUIREMENTS

3.1 (6626) **Reports: rocket engine and components**.- Report shall be prepared as follows:

3.1.1 (6626) General.-

3.1.1.1 (6626) Dimensional units.- Unless otherwise specified, all dimensional units shall be expressed in the English gravitational system of units.

3.1.1.2 (6626) Corrections.- Performance characteristics shall be corrected in accordance with the contractor's data reduction method. The performance characteristics shall be based on measurements obtained during a short time-interval at a stabilized point. The aforementioned conditions shall also be acceptable to the procuring activity.

3.1.1.3 (6626) Summary data sheets.-

3.1.1.3.1 (6626) Limits.- Performance limits and bench setting limits shall be superimposed on all summary curves. Performance limits shall be defined as the envelope of the curves which will give the rocket engine performance specified in this model specification.

3.1.1.3.2 (6626) Title block.- Each curve sheet or data plot shall contain the following information in a title block substantially in accordance with Figure 1 of MIL-E-6626A.

(a) Title (of summary)
(b) Component (nomenclature)
(c) Manufacturer (of component)
(d) Part No. (of component)
(e) Serial No. (of component)
(f) Test numbers (of original data sheets or curves from which summary curves are plotted)

3.1.1.3.2 (Continued)
(6626)

(g) Date

(h) Prepared by: approved by

(i) Contract No.

(j) Report No.

(k) Page No.

(l) Figure No.

(m) Contractor

(n) Testing activity

(o) Used on

3.1.1.3.3 (6626) Data block.- Test constants and other test data not recorded in the title block shall be recorded in a separate block substantially in accordance with Figure 1 of MIL-E-6626A.

3.1.2 (6626) Preliminary reports.- Immediately following completion of the engine tests and each separate component test, or consecutive group of tests conducted on any single test assembly or components, a brief report may be requested by the procuring activity. This report, combined with the certificate of a Government representative as to the proper conduct of the tests and the factual accuracy of the report may, at the discretion of the procuring activity, constitute the basis for approval of the tests.

3.1.2.1 (6626) Preparation.- Preliminary reports shall contain essentially the following information:

(a) General summary of test, giving dates, failures, test incidents, performance changes, marginal conditions, etc.

(b) Description of the condition of the engine or components, or both, at disassembly inspection.

3.1.2.1 (Continued)
(6626)

(c) Recommendations with respect to approval of the engine or components, or both, supplemented by such discussion as is necessary for their justification.

3.1.3 (6626) Final report.- Following completion of all tests required herein, a final report shall be submitted which will constitute a record of all information pertaining to the tests. This report will normally be used as a basis for approval of the Preliminary Flight Rating tests. The final report shall contain the following items.

3.1.3.1 (6626) Title page.-

3.1.3.2 (6626) Table of contents.-

3.1.3.3 Object.-

3.1.3.4 (6626) Summary.- (A brief summary of each of the tests conducted, giving the title of each test, the item tested, dates of testing, and a general statement of the results. References shall be made to the applicable preliminary reports.)

3.1.3.5 (6626) Conclusions and recommendations.-

3.1.3.6 (6626) Appendixes.- Each appendix shall cover a single test or group of consecutive tests conducted on any single test assembly or component, and shall report all test runs, and shall contain the following items.

3.1.3.6.1 (6626) (Brief general description of the rocket engine or of the components and a detailed description of all features which differ from the previous model, if applicable.)

3.1.3.6.2 (6626) Method of test (General description and schematic diagram of test equipment, methods, and measurement locations used in conducting the test).

3.1.3.6.3 (6626) Record of test (Chronological history of all events in connection with all the testing. The chronological history shall be presented in a graphical or equally acceptable form showing mid-point values of data as specified in paragraph (6626) 3.1.3.6.5.1.1.2. Failures, parts replacement, and other items of interest shall be noted.)

3.1.3.6.4 (6626) Analysis of results (A complete discussion of all phases of the tests, such as probable reasons for failure and unusual wear, comparison in performance with previous models, and analysis of general operation.)

3.1.3.6.5 (6626) Data.- Copies of specified data shall be furnished. (Where time functions are not originally recorded in curve form, the data shall be tabulated or plotted.)

3.1.3.6.5.1 (6626) Rocket Engine test.-

3.1.3.6.5.1.1 (6626) Specified data.- The following data may be furnished as rocket engine data for tests performed under missile simulated conditions, and as individual engine data under all other test conditions.

3.1.3.6.5.1.1.1 (6626) Original data.- Copies of original and reduced data shall be furnished for those cases in which transient effects are significant. In other cases, only reduced data shall be furnished. These data shall include but not be limited to the following:

(a) Thrust vs time
(b) Chamber pressure vs time
(c) Total oxidizer flow rate (single reading per data slice)
(d) Total fuel flow rate (single reading per data slice)
(e) External electrical power vs time
(f) Ambient air temperature (single reading per test)
(g) Propellant temperature at pump inlet (single reading per slice)
(h) Barometric pressure (single reading per test)

3.1.3.6.5.1.1.2 (6626) Derived data.- The derived data, based on data obtained at a stabilized point during a short time interval, shall include but not be limited to the following:

(a) Specific impulse
(b) Mixture ratio
(c) Thrust
(d) Chamber pressure
(e) Effective duration (not to be derived from short time interval data)

3.1.3.6.5.1.1.3 (6626) __Derived curves__.-

(a) Mixture ratio vs propellant utilization valve angle.

(b) Mixture ratio vs nominal chamber pressure.

3.1.3.6.5.1.2 (6626) __Test data__.- Test data shall include but not be limited to the following:

3.1.3.6.5.1.2.1 (6626) __Weight test__.- The weight of the rocket engine, determined during the acceptance test, shall be noted in the test report.

3.1.3.6.5.1.2.2 (6626) __Static leakage test__.- Not applicable.

3.1.3.6.5.1.2.3 (6626) __Drainage test__.- Not applicable.

3.1.3.6.5.1.2.4 (6626) __Vibration test__.- Wherever responses of components to the forcing vibrations are measured, the resonant frequencies and affected components shall be noted.

3.1.3.6.5.1.2.5 (6626) __Calibration test__.- Data as specified in paragraph (6626) 3.1.3.6.5.1.1 shall be furnished for the first, mid-point, and last test run.

3.1.3.6.5.1.2.6 (6626) __Variable thrust test__.- Not applicable.

3.1.3.6.5.1.2.7 (6626) __Safety limits__.- Data as specified in paragraph (6626) 3.1.3.6.5.1.1.1 shall be furnished where applicable for each malfunction test and for the first, mid-run, and the last start-shutdown test.

3.1.3.6.5.1.2.8 (6626) <u>Environmental test</u>.- Not applicable.

3.1.3.6.5.2 (6626) <u>Rocket engine component test</u>.-

3.1.3.6.5.2.1 (6626) <u>Power control test</u>.- Test data shall include but not be limited to the following.

3.1.3.6.5.2.1.1 (6626) <u>Altitude</u>.- Not applicable.

3.1.3.6.5.2.1.2 (6626) <u>Ignition-proof</u>.- Not applicable.

3.1.3.6.5.2.2 (6626) <u>Individual thrust-chamber assembly test</u>.- Test data shall include but not be limited to the following.

3.1.3.6.5.2.2.1 (6626) <u>Original data</u>.- Copies of original and reduced data shall be furnished for those cases in which transient effects are significant. In other cases, only reduced data shall be furnished. The data to be furnished shall be specified in this model specification. These data shall be sufficient to substantiate conformance with requirements for stable, safe, and reliable thrust chamber assembly operation.

3.1.3.6.5.2.2.2 (6626) <u>Derived curves</u>.- The following curves, in accordance with paragraph (6626) 3.1.1.2, shall be presented:

(a) Thrust vs chamber pressure
(b) Thrust coefficient vs chamber pressure
(c) Specific impulse vs chamber pressure

3.1.3.6.5.2.3 (6626) <u>Tank tests</u>.- Test data shall be furnished as required to demonstrate the capability of the tanks to meet service conditions.

3.1.3.6.5.3 (6626) <u>Other data</u>.- The Acceptance Test Log, as defined in paragraph (5152) 3.2.2 for the Preliminary Flight Rating test engine shall be included in the Preliminary Flight Rating test final report.

3.1.4 (6626) <u>Number and distribution of copies</u>.- Six copies of the Preliminary Flight Rating test report, one copy of which shall be reproducible, shall be forwarded to the procuring activity.

3.2 (6626) <u>Disposition of Preliminary Flight Rating test data</u>.- Preliminary Flight Rating test data shall be retained by the contractor for two years and furnished to the procuring activity or authorized representative upon request.

4. (6626) QUALITY ASSURANCE PROVISIONS

4.1 (6626) <u>General</u>.- Liquid propellant rocket engines, components, and test apparatus shall be subject to inspection by authorized Government Inspectors. All tests outlined in this specification shall be subject to witnessing by representatives of the contractor and the procuring activity. Two copies of the complete parts list and specifications for all components of the flight rating test engine shall be furnished to the procuring activity prior to beginning the Preliminary Flight Rating tests. The drawings, prior to and during the Preliminary Flight Rating tests, shall be at the disposal of the

4.1 (Continued)
(6626)

authorized Government Inspectors. At convenient times prior to the tests and after the tests, the rocket engine and components shall be examined to determine if they conform to all requirements of the contract and specifications under which they were built. At the option of the procuring activity, measurements shall be made of critical engine dimensions prior to start of the Preliminary Flight Rating tests. During the progress of tests, examinations may be made at the option of the procuring activity. The results of all such examinations shall be submitted as part of the Preliminary Flight Rating test data.

4.1.1
(6626)

<u>Test apparatus and procedures</u>.- Schematic drawings and descriptions of all test apparatus and outline diagrams showing points of the measuring apparatus and its application shall be furnished prior to initiation of the Preliminary Flight Rating test. The plumbing runs to the YLR101-NA-13 or YLR101-NA-15 shall simulate the missile installation. Any deviations from missile installation simulation shall be approved by the procuring activity. Test procedures and methods to be used shall be acceptable to the procuring activity.

4.1.1.1
(6626)

<u>Instrument calibration.</u>- Each instrument and other measuring apparatus upon which the accuracy of test results depends shall be calibrated frequently enough to ensure attainment of steady-state accuracy of plus or minus 3 percent of the specified value of measurement, except where greater accuracy

4.1.1.1 (Continued)
(6626)

is required to demonstrate the model specification requirements. Calibration records shall be maintained and shall be made available to authorized representatives of the procuring activity or of the contractor upon request.

4.1.1.2 (6626) <u>Automatic recording equipment</u>.- Automatic recording equipment of adequate response shall be used to obtain data during transient conditions of engine and component operation requiring the evaluation of time versus engine variables.

4.1.2 (6626) <u>Test conditions.</u>- Unless otherwise specified, all inspections and tests shall be conducted at room temperature, as defined in paragraph (6626) 4.1.2.1.2 entitled "Room temperature", and at ambient pressure. The rocket engine shall be tested with the pump inlet conditions with the limits specified in paragraph 3.6.1.1. Engine purges used in the tests shall simulate the engine purges as required for missile captive tests. Ground support equipment, consisting of ground electrical control console, and ground electrical box or acceptable equivalent shall be used to simulate operational checkout procedures as closely as possible, while maintaining maximum information of engine condition prior to each test firing.

4.1.2.1 (6626) <u>Temperatures</u>.-

4.1.2.1.1 (6626) <u>Low temperatures</u>.- Not applicable.

4.1.2.1.2 (6626) Room temperature.- Room temperature limits are as follows: (a) Fuel at plus 32 to plus 80 F, (b) hydraulic fluid at plus 50 to 110 F, (c) ambient air and other test fluids at plus 10 to plus 110 F, except:

(1) When the test fluid boiling point is below plus 10 F the fluid shall be allowed to remain at the temperature encountered under test conditions.

(2) When the test fluid freezing point is above plus 10 F, the fluid shall be maintained at 10 F above its freezing point unless otherwise specified in the model specification.

4.1.2.1.3 (6626) High temperature.- Not applicable.

4.1.2.1.4 (6626) Turbine drive fluid temperature.- Fluids supplied for turbine drives shall be measured.

4.1.3 (6626) Parts failure and replacement.- Maintenance, adjustment, or replacement of parts other than described to preflight check-out shall not be permitted during testing, except as mutually agreed upon between the contractor and the procuring activity.

4.1.3.1 (6626) Rocket engine.- If, during the Preliminary Flight Rating test of the engine, a part fails, this part may be replaced or a new Preliminary Flight Rating test started on a new engine with a redesigned part or one of different material, unless the procuring activity authorizes the installation of a new part of original design and material for one which, in the judgement of the procuring activity, failed due to faulty material or workmanship. The Preliminary Flight Rating test on the engine shall be considered complete when every part of the engine has been subjected to, and has satisfactorily completed an entire test. At the discretion of the procuring activity, redesign and retesting may be required of any part which fails or indicates weakness after completing its Preliminary Flight Rating test but is retained in the engine to complete testing on other parts.

4.1.3.2 (6626) Components.- The above procedure shall apply in the event of parts failure during the flight rating testing of components.

4.2 (6626) Rocket engine inspections and tests.- The rocket engine to be subjected to a Preliminary Flight Rating test shall be a deliverable system, which shall be demonstrated satisfactorily by the performance of the acceptance tests of Specification MIL-E-5152A as modified by paragraph 4.2.4.

4.2.1 (6626) Rocket engine tests.- The rocket engine shall be subjected consecutively to the drainage, static leakage, vibration, calibration, and static leakage tests. Safety limits tests may be scheduled as convenient and may be performed on another identical engine which shall be subjected consecutively to the static leakage, the safety limits tests, and static leakage tests.

4.2.1 (Continued)
(6626)

Unless otherwise specified herein, any additional tests required by the procuring activity under paragraph (6626) 4.2.1.9 shall be conducted after the safety limits tests. A minimum running time shall be accumulated, prior to the final static leakage tests, equal to 6 runs at rated duration, excluding the time accumulated during the safety limits tests. At least three (3) runs shall be for rated duration. Rated duration shall be considered to be the duration for the complete sequence of engine operation including the vernier solo duration.

4.2.1.1 (6626) Weight.- The dry rocket engine shall be weighed, and its weight shall not exceed the value specified in the model specification.

4.2.1.2 (6626) Static leakage.- All fluid systems of the rocket engine, as specified in paragraph 3.6.1, shall be tested for leakage by pressurizing individual systems to full operating pressure wherever practicable or to the highest pressures technically feasible considering recognized safety factors. The test pressure shall start at a low differential pressure and be increased at a uniform rate to the static leakage pressure. The maximum test pressure shall be maintained for a minimum of 2 minutes. Leakage, at any time during the test, shall not exceed that specified in paragraph 3.3.9.5, drawings, and component specifications.

4.2.1.3 (6626) Drainage.- The fluid systems of the engine shall be completely filled, with the engine in a vertical position, then drained and purged to the maximum extent possible without firing. The fluids remaining shall be determined and shall not exceed the amounts specified in the model specification.

4.2.1.4 (6626) Vibration.- The vibration test shall consist of the calibration test runs of paragraph (6626) 4.2.1.5. Evidence of fatigue failure of any component upon completion of these tests shall be cause for rejection of the component.

4.2.1.5 (6626) Calibration.- Data shall be measured and calculated for the following tests in accordance with paragraph (6626) 3.1.3.6.5.1.2.5. The rocket engine shall be in a vertical position and the sequence of its operation shall be as specified in paragraphs 3.3.5 and 3.3.6.

4.2.1.5.1 (6626) Calibration tests.- During the calibration tests, measurements shall be taken that demonstrate that the rocket engine meets the following requirements: (a) the cutoff impulse, as specified in paragraph 3.3.10.2.3, and (b) the ratings of Table I of paragraph 3.3.2. During one duration calibration test, the head suppression valve (paragraph 3.6.10.1) shall be operated with the inlet head to the rocket engine varied within the range and rates expected from the missile propellant delivery system, subject to the limits of the test facility. The propellant utilization valve shall be operated to demonstrate the mixture ratio limits of paragraph 3.9.1 during one (1) calibration test.

4.2.1.5.1.1 (6626) Calibration, gimbaling fixed.- Two tests shall be made at a thrust rating within the range of Table I, of paragraph 3.3.2, with the thrust chamber in the fixed position, one of which shall be at rated duration.

4.2.1.5.1.2 (6626) Calibration, gimbaling operative.- Two (2) runs shall be made at rated duration at a thrust rating within the range of Table I, paragraph 3.3.2, during which time the gimbaling shall be operated throughout the requirements of paragraph 3.6.6.

4.2.1.5.2 (6626) Variable and multiple thrust ratings.- Not applicable.

4.2.1.6 (6626) Variable thrust.- Not applicable.

4.2.1.7 (6626) Safety limits.- A total of 20 tests shall be performed on the rocket engine. They shall consist of a sufficient number of tests to meet the requirements of paragraph (6626) 4.2.1.7.1, and the remainder shall be in accordance with paragraph (6626) 4.2.1.7.2. During any five of these tests, the rocket engine shall be operated at the high and low voltage limit for starting sequence and mainstage operation as specified in paragraph 3.3.8 (b). Two (2) tests shall be conducted at the high limit and three (3) at the low limit of the voltage range.

4.2.1.7.1 (6626) Malfunction.- The type and order of these tests shall be based on the malfunction analysis required by paragraph 3.3.7, entitled "Malfunction", and shall be included with the tests specified in paragraph (6626) 4.2.1.7 as described by the approved test procedure. Compliance with paragraph 3.3.7 shall be demonstrated after the occurence of one of the following events during either transient or stabilized operation of the rocket engines:

(a) Malfunction of rocket engine component or system.
(b) Malfunction of the vehicle system affecting rocket engine operation.

4.2.1.7.2 (6626) Start-shutdown.- The number of those tests shall be sufficient to complete the total number of tests required by paragraph 4.2.1.7. The rocket engine shall be subjected consecutively to the following:

(a) Starting cycle and sequence of events outlined in paragraph 3.3.5.

(b) Stabilized operation at a level and of sufficient duration to obtain data substantiating the requirements for stable, safe, reliable operation.

(c) Shutdown cycle and sequence of events outlined in paragraph 3.3.6.

4.2.1.8 (6626) Environmental.- Not applicable.

4.2.1.9 (6626) Additional tests.- The procuring activity may require additional tests for the purpose of testing special features of the rocket engine and propellants. These tests shall be as required and mutually agreed upon by the contractor and the procuring activity, and in general, shall not increase the total running time accumulated during the Preliminary Flight Rating test.

4.2.1.10 (6626) Preliminary Flight Rating conditions.- Preliminary Flight Rating of the rocket engine shall be predicated on maintenance of all parameters within the limits and conditions specified herein, except for the maintenance of those parameters restricted by the missile simulated configuration. Minor parts failures or malfunctions may, at the option of the procuring activity, be considered acceptable if safety is not jeopardized. The failures or malfunctions shall be recorded in accordance with paragraph (6626) 3.1.3.6.3 and submitted to the procuring activity for approval.

4.2.2 (6626) Rocket engine inspection after test.- After completion of tests of the rocket engine, the engine shall be completely disassembled for examination of all parts, measurements, and photographs taken as necessary to disclose excessively worn, distorted, or weakened parts. Calibrations shall be made of all controls and control components prior to disassembly. These calibrations shall demonstrate the components are within the design tolerance range required by the applicable specification.

4.3 (6626) Component inspection and tests.-

4.3.1 (6626) Previous component qualification.- All rocket engine components requiring flight rating inspection and test as specified herein may have these requirements waived at the option of the procuring activity, if the component has been previously qualified or has passed Preliminary Flight Rating tests at the same or higher rating for service use on another engine. The components must be substantially identical to the respective components previously qualified or flight rated with the exception of provisions for engine installation. If such a waiver is granted, informat on on the components for which previous approval was obtained shall be provided in the Preliminary Flight Rating test report.

4.3.2 (6626) Component inspection before tests.- All components shall be completely inspected for compliance with the contractor's drawings and specifications before Preliminary Flight Rating tests are begun. Deviations from the contractor's drawings and specifications shall be approved by a representative of the procuring activity. Defective parts shall not be used on any component or engine subjected to the Preliminary Flight Rating tests.

4.3.3 (6626) Component tests.- The following tests shall be conducted on components.

4.3.3.1 (6626) Power control tests.- Each electrical or altitude sensitive subcontrol shall be tested in accordance with the following paragraphs. A functional test shall simulate as closely as possible the entire range encountered in engine operation.

4.3.3.1.1 (6626) Altitude.- Pressure altitudes shall be as defined in the May 1954 edition of the Manual of the ICAO Standard Atmosphere - Calculations by FAA (formerly NACA) and the April 1955 draft of Appendix A, entitled, "Proposed Extension to the ICAO Standard Atmosphere, Model Ib".

4.3.3.1.1.1 (6626) Pressure sensitive subcontrol test.- Each subcontrol, sensitive to altitude, with its operating fluid or approved test fluid, shall be functionally tested for 10 cycles at a simulated pressure altitude of 200,000 feet.

4.3.3.1.1.2 (6626) Electrical subcontrol test.- Each electrical subcontrol, with its operating fluid or approved test fluid, shall be subjected to the following:

(a) The test chamber pressure shall be established at the pressure existing at a pressure altitude of 200,000 feet.

(b) Electrical power shall be applied for 10 minutes with all power sources at maximum design values.

(c) A functional test of 100 cycles, in accordance with Specification MIL-E-5151, shall be performed while the subcontrol is operated over the input range of either 18 to 29 volts dc or 102 to 124 volts ac throughout the frequency range required by the type of ac power utilized as defined in Specification MIL-E-7894.

4.3.3.1.1.2 (Continued)
(6626)

(d) While the subcontrol is hot as a result of the functional test, a potential of 300 volts rms for relays and solenoids at commercial frequency, shall be applied between all terminals not in the same circuit and between terminals and grounded metal parts for a period of 60 seconds.

There shall be no "arc-over" or evidence of "arc-over" between electrical contacts, terminals, or parts of a subcontrol having a difference of potential. Current flow in excess of 2 milliamperes or breakdown of insulation shall constitute failure.

4.3.3.1.2 (6626) <u>Ignition-proof</u>.- Electrical subcontrols, except those specified in paragraph 4.3.6 of Specification MIL-E-5272, shall be tested in accordance with paragraph 4.13.1, Explosion Proof (Aeronautical) test, Procedure I, of that specification. The altitude increments shall be 10,000 feet and shall be from sea-level to 60,000 feet. Components which operate only at take-off shall be tested at sea level and 10,000 feet altitudes only.

4.3.3.2 (6626) <u>Individual thrust chamber assembly tests</u>.-

4.3.3.2.1 (6626) <u>Injector calibration.</u>- Not applicable.

4.3.3.2.1.1 (6626) <u>Coolant jacket calibration</u>.- Not applicable.

4.3.3.2.2 (6626) Calibration.- The thrust chamber assembly shall be calibrated at 90 percent rated thrust, 110 percent rated thrust, and one intermediate thrust. Two calibrations shall be made at each level. Standard thrust chamber assembly calibration curves, showing relationships of propellant flow rate, thrust, and thrust coefficient with chamber pressure shall be prepared from the test results. These curves, together with the best available data, shall be used to determine thrust from the chamber pressure whenever direct means of measuring thrust are not available or feasible.

4.3.3.2.3 (6626) Acceptance conditions.- Acceptance of each thrust-chamber assembly shall be predicated on the maintenance of all parameters within the limits specified in this model specification.

4.3.3.3 (6626) Tank tests.- Each type of tank, which is supplied with the rocket engine shall be subjected to proof-pressure test, pressure cycling, and burst-pressure tests. Minimum burst-pressure shall be not less than 1.33 times proof-pressure, and pressure cycling shall be conducted for 500 cycles between approximately zero psig and working pressure. The test requirements of Specification MIL-T-5208A are not applicable.

4.3.4 (6626) Component inspection after tests.- After completion of tests of the components, each component shall be completely disassembled for examination of all parts and measurements, as necessary, to disclose excessively worn, distorted, or weakened parts. These measurements shall be compared with the contractor's drawing dimensions and tolerances or with similar measurements made prior to the test when available. The results of these inspections shall be submitted as part of the Preliminary Flight Rating test data.

5. (6626) PREPARATION FOR DELIVERY (Not applicable to this specification)

6. (6626) NOTES

6.1 (6626) Intended use.- The Preliminary Flight Rating test procedures specified herein are intended for use in the testing of liquid-propellant rocket engines.

6.2 (6626) Definitions and symbols.- The symbols and terms used in this specification and the applicable definitions are as specified in Specification MIL-E-5150A as modified herein.

4.2.4 Acceptance tests.- The Acceptance tests shall be conducted on each engine in accordance with Specification MIL-E-5152 except as modified or reiterated hereafter with paragraph numbers identified by (5152).

1. (5152) SCOPE

1.1 (5152) This specification covers the Acceptance test requirements for the rocket engine.

2. (5152) APPLICABLE DOCUMENTS

2.1 (5152) The applicable publications listed in the following bulletin, of the issue specified in the manufacturer's model specification, form a part of this specification:

Air Force - Navy Aeronautical Bulletin

No. 343 Specifications and Standards Applicable to Aircraft Engines and Propellers: Use of

(Copies of specifications, standards, drawings, and publications required by contractors in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.)

3. (5152) REQUIREMENTS

3.1 (5152) Contractor's instructions, specifications and drawings.-

3.1.1 (5152) Specifications.- Contractor's specifications shall include any tests found necessary to ensure calibration of components within specified environmental and operating conditions regardless of manufacturing processes employed.

3.1.2 (5152) Component calibration.- A complete set of calibration specifications for the calibrated components showing the bench setting limits used for acceptance shall be made available to the procuring activity prior to delivery of the first rocket engine.

3.1.3 (5152) Availability.- Copies of all contractor instructions, specifications, and drawings governing the inspection and testing of the rocket engine and its components shall be kept on file by the contractor and shall be supplied to the procuring activity upon request. When instructions, specifications, or drawings have been supplied to the procuring activity, all changes shall be supplied to the procuring activity upon release.

3.2 (5152) Acceptance test data.-

3.2.1 (5152) General.-

3.2.1.1 (5152) Dimensional units.- Unless otherwise specified, all dimensions shall be reported in the English gravitational system of units.

3.2.1.2 (5152) Corrections.- Performance characteristics shall be corrected in accordance with the contractor's data reduction method which shall be based on measurements obtained at a stabilized point during a short time-interval and which shall be acceptable to the procuring activity.

3.2.2 (5152) Log.- An Acceptance test log shall be prepared for each engine to include but not be limited to the reduced and derived data listed below.

3.2.2.1 (5152) Component test and inspection data files.- Data shall include test conditions and serial numbers of components tested, so arranged that summaries of tests on specific components by groups of serial numbers can be compiled.

3.2.2.2 (5152) Individual thrust chamber assembly.- Not applicable.

3.2.2.3 (5152) Rocket engine.-

YLR105-NA-7.-

(a) Weight tests
(b) Static leakage tests
(c) For each calibration test, a continuous record shall be taken of the original data for each engine.
   (1) Total thrust (when measured directly)
   (2) Total fuel flow rate
   (3) Total oxidizer flow rate
   (4) Chamber pressure (for each thrust chamber)
   (5) Barometric pressure and free air temperature (single reading)
   (6) Temperature of propellants at inlet to rocket engine
   (7) Fuel pump inlet pressure
   (8) Oxidizer pump inlet pressure
   (9) YLR105-NA-7 pump bleed flow rates for the YLR101-NA-15.
(d) Additional tests.- (Use schedule for paragraph 3.2.2.3.2 (c).)

3.2.2.4 (5152) Reduced and derived data.-

3.2.2.4.1 (5152) Individual thrust chamber assembly.- For each run the minimum data based on data obtained at a stabilized point during a short time interval shall include the following:

(a) Instantaneous specific impulse
(b) Instantaneous mixture ratio
(c) Instantaneous thrust
(d) Instantaneous chamber pressure

3.2.2.4.2 (5152) Rocket engine.- For each run the minimum data shall include the following:

(a) Instantaneous specific impulse
(b) Instantaneous mixture ratio
(c) Inlet temperature of propellant

3.2.2.5 (5152) Inspections.- Results of the specified rocket engine inspections shall be included in the Acceptance test log.

3.2.2.6 (5152) Disposition.- The Acceptance test log shall be retained by the contractor for two (2) years, and copies shall be furnished to the procuring activity upon request.

4. (5152) QUALITY ASSURANCE PROVISIONS

4.1 (5152) General.- Liquid propellant rocket engines, components, and test apparatus and the material entering into the manufacture of articles for fulfillment of contract requirements, shall be subject to inspection by authorized Government Inspectors. Complete specification for all components shall be furnished to the procuring activity prio to beginning the Acceptance tests thereof. At convenient times prio to the tests and after the tests, the rocket engine and components shall be examined to determine if they conform to all requirements of the contract and specifications under which they were built. During the progress of tests, examinations may be made at the option of the procuring activity.

4.1.1 (5152) Test apparatus and procedures.- Schematic drawings and descriptions of all test apparatus, and outline diagrams showing points of measuring apparatus, application, and test procedures or methods to be used shall be acceptable to the procuring activity.

4.1.1.1 (5152) Instrumentation calibration.- Each instrument and other measuring apparatus upon which the accuracy of test results depends shall be calibrated frequently enough to ensure attainment of steady state accuracy of plus or minus 3 percent of the specified value of the measurement, except where greater instrumentation accuracy is required to demonstrate the specification requirements. Calibration records shall be maintained and shall be made available to authorized representatives of the procuring activity or of the contractor upon request.

4.1.1.2 (5152) Automatic recording equipment.- Automatic recording equipment of adequate response shall be used to obtain data during transient conditions of rocket engine and component operation requiring the evaluation of time versus rocket engine variables.

4.1.3 (5152) Test temperatures.- Inspections and tests shall be conducted at room temperature as defined in paragraph (5152) 4.1.3.1 and at ambient pressure.

4.1.3.1 (5152) Room temperature.- Room temperature limits are as follows: (a) fuel at plus 32 to plus 80 F, (b) hydraulic fluid at plus 50 to plus 110 F, (c) ambient air and other test fluids at plus 10 to plus 110 F, except:

4.1.3.1 (Continued)
(5152)

(1) When the test fluid boiling point is below plus 10 degrees F, the fluid shall be allowed to remain at the temperature encountered under test conditions.

(2) When the test fluid freezing point is above plus 10 degrees F, the fluid shall be maintained at 10 degrees F above its freezing point unless otherwise specified in the model specification.

4.1.3.2 (5152) Turbine drive fluid temperature.- Not applicable.

4.2 (5152) Acceptance tests.- Unless otherwise specified herein, the Acceptance tests shall be conducted on each production rocket engine and shall consist of the tests specified under Schedule "A" or "B". All production rocket engines shall be acceptance-tested under Schedule "A" until such time as the penalty or parts replacement record warrents the use of Schedule "B" as mutually agreed upon by the contractor and the procuring activity. All subsequent rocket engines, except one out of a lot to be agreed upon between the contractor and the procuring activity, shall be acceptance-tested in accordance with Schedule "B". This rocket engine, selected at random by the Government Inspector, shall be acceptance-tested in accordance with Schedule "A". If the rocket engine does not include its own feed system, the rocket engine shall be tested with a test stand feed system providing inlet conditions as specified herein.

4.2.1 (5152) Rocket engine inspection before acceptance tests.- Each rocket engine shall be completely assembled in accordance with the contractor's drawings and this specification then visually and dimensionally inspected before commencing the rocket engine tests.

4.2.2 (5152) Rocket engine tests.- Schedule "A".- Each rocket engine assembled for the inspection specified in paragraph (5152) 4.2.1 shall be subjected to the weight, static leakage, calibration, and such additional tests as required by paragraph (5152) 4.2.2.6. Accessory equipment listed in paragraph 3.6.10 shall not be required to be acceptance-tested with the engines. Acceptance of the propellant tanks and associated components may be accomplished independently.

4.2.2.1 (5152) Weight.- The dry rocket engine shall be weighed, and its weight shall not exceed the value specified in the model specification.

4.2.2.2 (5152) Radio interference.- Not applicable.

4.2.2.3 (5152) Static leakage.- All fluid systems of the rocket engine shall be tested for leakage by pressurizing individual systems to full operating pressures wherever practicable or to the highest pressures technically feasible or limited by recognized safety factors. The test pressures shall start at a low differential pressure and be increased at a uniform rate to the static leakage pressure. The maximum test pressure shall be maintained for a minimum of 2.0 minutes. Leakage, at any time during the test, shall not exceed that specified in paragraph 3.3.9.5.

4.2.2.4 (5152) Calibration.- Calibration hot firing shall be conducted as follows: The engines shall be operated for a minimum duration to demonstrate compliance with the performance ratings of Table I of paragraph 3.3.2. YLR101-NA-15 Bleeds shall be simulated.

4.2.2.5 (5152) Rocket engine component tests.- All component parts of the rocket engine shall be tested to demonstrate conformance with the contractor's drawings and specifications.

4.2.2.5.1 (5152) Individual thrust chamber assembly tests.-

4.2.2.5.1.1 (5152) Flow tests.- Each individual thrust chamber assembly shall be given a cooling jacket flow test and an injector flow test using the propellants specified in the model specification or an alternate test fluid approved by the procuring activity in accordance with approved test procedures. The pressure drops, at rated flow conditions, shall be within the limits specified in the calibration curves called for in paragraph 3.2.1 (5152).

4.2.2.5.1.2 (5152) Firing tests.- Calibration tests performed under paragraph (5152) 4.2.2.4 shall accomplish the requirements for individual thrust chamber firing tests.

4.2.2.5.2 (5152) Gas generator test.- Gas generator calibration shall be accomplished under paragraph (5152) 4.2.2.4 with the gas generators assembled with the engine.

4.2.2.6 (5152) Additional tests.- Additional tests, for the purpose of testing special features of the rocket engine and propellants, shall be as required by mutual agreement of the procuring activity and the contractor. These tests shall not, in general, increase the total running time accumulated during the acceptance test.

4.2.3 (5152) Rocket engine and component tests.- Schedule "B".- Schedule "B" shall consist of the test requirements agreed upon between the procuring activity and contractor for the rocket engine and components involved.

4.2.4 (5152) Acceptance conditions.- Acceptance of the rocket engine shall be predicated on the maintenance of all parameters within the limits specified in Table I of Paragraph 3.3.2 when corrected to the nominal rated mixture ratio. Stable operation shall be demonstrated throughout the entire mixture ratio range of 2.27 plus or minus 15 percent. YLR101-NA-15 propellant bleeds shall be simulated. Following the final acceptance run each engine shall be adjusted to target nominal rated thrust by adjusting the power regulator. An average power regulator setting shall be used as determined from acceptance test runs. The final setting shall be entered in the engine acceptance log. The average specific impulse of each engine, determined from the acceptance runs, shall be above the specification minimum. An individual test value below the minimum specific impulse but within 2 percent of the average shall be acceptable.

4.2.5 (5152) Rocket engine inspection after test.- Upon completion of the acceptance tests, the rocket engine shall be subject to a complete visual inspection by the Government Inspector, without disassembly, except where data or circumstances indicate that defective parts may exist; the disassembly may be accomplished as requested by the Government Inspector. Defective parts shall be replaced by approved parts, and a suitable penalty test may be made at the discretion of the Government Inspector if the replaced parts failed under unusual circumstances or affect performance characteristics of the engine.

4.2.5.1 (5152) Rocket engine penalty test.- The maximum penalty test shall consist of a repetition of the test runs outlined under paragraph (5152) 4.2.2. Preliminary runs may be conducted prior to the penalty test.

4.2.5.2 (5152) Rocket engine inspection after penalty test.- Upon completion of the acceptance test the rocket engine shall be subjected to a complete visual inspection by the Government Inspector without disassembly except where data or circumstances indicate that defective parts may exist; then, disassembly may be accomplished as requested by the Government Inspector. Defective parts shall be replaced by approved parts and a suitable penalty test may be made at the discretion of the Government Inspector if replaced parts affect operational characteristics of the engine.

4.2.5.3 (5152) Rocket engine assembly test.- Not applicable.

4.2.6 (5152) Rejection and retest.- Whenever, in the opinion of the inspector, there is evidence of malfunction or that the rocket engine is not meeting performance rating requirements, the difficulty shall be investigated and its cause corrected to the satisfaction of the inspector before the test is continued. At the option of the inspector, the portion of the test in which the difficulty was encountered shall be repeated.

4.2.6.1 (5152) Radio interference.- Not applicable.

4.2.6.2 (5152) Maximum running time.- The rocket engine shall stand rejected whenever the total running time accumulated during preliminary runs and the tests specified herein exceeds three (3) times rated duration. Parts and accessories from rejected rocket engines may be reused if such items can be reconditioned to meet the requirements for new parts. The inspector shall be furnished full particulars of previous rocket engine rejections when such items are resubmitted for inspection.

5. (5152) PREPARATION FOR DELIVERY

5.1 (5152) Not applicable to this specification.

6. (5152) NOTES

6.1 (5152) Symbols and definitions.- The symbols and terms used in this specification and the applicable definitions will be as specified in this model specification.

4.2.4.1 Miscellaneous inspection test.-

4.2.4.1.1 Material tests.- Samples of materials used in the rocket engine shall be selected and tested as specified in the quality control procedures established by the contractor.

4.2.4.1.2 Magnetic inspection.- The following parts shall be subjected to magnetic particles inspection in accordance with Specification MIL-I-6868 or AMS 2640, if made of magnetic material.

(a) All highly stressed parts constituting the pump-turbine rotor assembly, including threaded fastenings.
(b) Other highly stressed parts.
(c) Vibration or friction damper springs.
(d) All gears.

4.2.4.1.3 Fluorescent penetrant inspection.- The following nonmagnetic parts shall be subjected to fluorescent penetrant inspection in accordance with Specification MIL-1-6868 or AMS 2645:

(a) Pump impellers
(b) Turbine blades and rotor
(c) All other highly stressed parts

4.2.4.1.3.1 Very bulky and intricately shaped parts may be hydrostatically tested by the contractor's approved method in lieu of fluorescent penetrant testing, when specifically approved by the procuring activity's representative.

4.2.4.1.4 Utility parts.- Commercial, AN, and MS standard parts such as cotter pins, washers, etc., and similar low-stressed parts are not required to be inspected by the magnetic or fluorescent penetrant method.

4.2.4.1.4.1 Antifriction bearings.- Assembled ball or roller bearings shall not be magnetically inspected.

4.2.4.1.5 Radiographic or ultrasonic inspection.- The following shall be subjected to radiographic or ultrasonic inspection for defects or soundness to a degree of inspection on each article as agreed upon between the contractor and the procuring activity:

(a) The propellant pump impeller(s), or rotor(s), if nonmagnetic.
(b) The turbine rotor(s), if nonmagnetic.
(c) Highly stressed magnesium and aluminum castings.

4.2.4.1.5.1 Radiographic inspection.- Radiographic inspection materials shall be in accordance with Specification MIL-I-6865. Laboratories performing radiographic inspection shall be certified in accordance with Specification MIL-X-6141.

4.2.4.1.6 Certification of operators.- All operators performing fusion welding shall be certified.

5. PREPARATION FOR DELIVERY

5.1 Application.- The requirements of Section 5 apply only to direct purchases by or direct shipment to the Government.

5.2 Storage, shipment, and delivery.- Preparation for storage and shipment, when required by contract, shall be in accordance with the applicable specifications and drawings of paragraph 3.5, Drawings and data, utilizing government-approved containers unless otherwise directed by the contracting officer or his authorized representative.

6. NOTES

6.1 Intended use.- The liquid propellant rocket engines covered by this specification are intended for remotely launched missile application.

6.2 Symbols and definitions.- The symbols used in the model specifications, and the applicable definitions, will be as specified in Specification MIL-E-5150 except as follows:

6.2.1 Definitions.-

6.2.1.1 Government.- The term "Government" as used in this specification shall be interpreted to mean the procuring activities of the Department of Defense.

6.2.1.2 Procuring activity.- The procuring activity is the activity which negotiates the rocket engine contract.

6.2.1.3 Using service.- The using service is the activity whose model dash number has been assigned to the rocket engine in accordance with ANA Bulletin No. 352.

6.2.1.4 Rating.- A value of some characteristic of performance as specified in the model specification.

6.2.1.5 Estimate.- A predicted range of characteristics of performance as specified in the model specification.

6.2.1.6 Accessories.- Accessories are items of equipment required for vehicle operation.

6.2.1.7 Assembly, pump and drive.- The pump and drive assembly consists of the propellant pump(s), pump drive(s), gas generator (if used), and all necessary controls.

6.2.1.8 Assembly, thrust chamber.- The thrust chamber assembly (TCA) is composed of the thrust chamber, and any other directly associated parts.

6.2.1.9 Chamber, combustion.- The combustion chamber is the enclosed volume between the injector face and the throat of the nozzle.

6.2.1.10 Chamber, thrust.- The thrust chamber is that component of rocket engine which produces thrust and includes the expansion nozzle and propellant injector. (Propellant valves are included if they are an integral part of the injector).

6.2.1.11 Coefficient, thrust.- The thrust coefficient (C_F) is the quotient of the thrust in pounds divided by the product of the nominal chamber pressure in pounds per square inch absolute and the throat area in square inches.

6.2.1.12 Components, engine.- Engine components are items of equipment furnished as parts of the engine which are required for engine operation.

6.2.1.13 Consumption, specific propellant.- The specific propellant consumption (SPC) is the total propellant consumption rate, in pounds per second, divided by the thrust produced, in pounds.

6.2.1.14 Conditions, standard.- Standard conditions are the values of air temperature and pressure given in Appendix A, Proposed Extension to ICAO Standard Atmosphere, Model 1b, Using Variable Gravity, Molecular-Scale Temperatue and Geopotential Altitude, AFRC-ARDC, dated April 1955.

6.2.1.15 Cutoff.- Cutoff is the time of propellant flow cessation through the thrust chamber propellant shutoff valve(s).

6.2.1.16 Drive, pump.- The pump drive consists of any power source and necessary controls, to operate the propellant pumps.

6.2.1.17 Duration.- The duration is the total firing time of one operational cycle (seconds).

6.2.1.18 Efficiency, over-all pump.- The over-all pump efficiency is the ratio of the (Hydraulic) output horsepower to the input horsepower.

6.2.1.19 Efficiency, turbine mechanical.- The turbine mechanical efficiency is the ratio of the shaft horsepower to ideal isentropic energy available by expansion from the inlet total temperature and total pressure to the outlet static pressure.

6.2.1.20 Head, net, positive suction.- The net positive suction head (NPSH) is the total absolute pump-inlet pressure above fluid-vapor pressure, expressed in feet of fluid.

6.2.1.21 Impulse, effective.- Effective impulse is the area under the thrust-time curve between the two 90-percent-of-rated thrust points.

6.2.1.22 Impulse, effective specific.- The effective specific impulse is the effective impulse divided by the sum of the weights of propellants used during the intervals between the two 90-percent-of-rated thrust points.

6.2.1.23 Impulse, instantaneous specific.- The instantaneous specific impulse is the instantaneous thrust produced, in pounds, divided by the total instantaneous propellant consumption rate, in pounds per second.

6.2.1.24 Impulse, instantaneous specific.- The instantaneous specific impulse is the instantaneous thrust produced, in pounds, divided by the total instantaneous propellant consumption rate, in pounds per second.

6.2.1.25 Impulse, total.- The total impulse (I_t) is the area under the thrust-time curve.

6.2.1.26 Length, characteristic.- The characteristic length (L^*) is the combustion chamber volume in cubic inches divided by the throat area in square inches.

$$L^* = \frac{V_c}{A_t} \text{ inches}$$

6.2.1.27 Points, 90-percent-of-rated thrust.- The 90-percent-of-rated thrust points are the time points during thrust increase and decrease between which the thrust is stabilized at greater than 90 percent of rated value.

6.2.1.28 Pressure, effective chamber.- The effective chamber pressure is the area under chamber pressure-time curve between the two 90-percent-of-rated thrust points divided by the time interval between these points.

6.2.1.29 Pressure, mean chamber.- The mean chamber pressure is the area under the chamber pressure-time curve divided by the duration.

6.2.1.30 Pressure, nominal working.- The nominal working pressure is a maximum pressure to which the component is subjected under steady state conditions.

6.2.1.31 Pressure, burst.- Burst pressure is the pressure which, once applied to an item, results in exceeding its ultimate strength.

6.2.1.32 Pressure, proof.- Proof pressure is the test pressure to which an item is subjected without deformation adversely affecting rocket engine operation, or permanent set. Proof pressure is 1.5 times the nominal working pressure for aircraft- and vehicle-launched rocket engines and 1.2 times the nominal working pressure for remotely launched missile rocket engines plus the difference between nominal working pressure and maximum transient pressure.

6.2.1.33 Pressure, maximum transient.- The maximum transient pressure is the significant maximum pressure to which an item is subjected under any operating condition.

6.2.1.34 Propellant, referee.- A propellant incorporating the most adverse constituents of the specification propellant or which specified propellant constituents after a 2-year storage period.

6.2.1.35 Ratio, effective mixture.- The effective mixture ratio is the weight of the oxidizer used between the two 90-percent-of-rated thrust points divided by the weight of the fuel used between the two 90-percent-of-rated thrust points.

6.2.1.36 Ratio, mean mixture.- The mean mixture ratio (W_o/W_f) is the total weight of oxidizer consumed divided by the total weight of fuel consumed.

6.2.1.37 Ratio, instantaneous mixture.- The instantaneous mixture ratio (r_m) is the ratio of the oxidizer flow rate to the fuel flow rate.

6.2.1.38 Rocket engine.- A rocket engine consists of all components specified in the model specification.

6.2.1.39 Rocket engine, aircraft.- "Aircraft" denotes a propulsion rocket engine for an inhabited flight vehicle.

6.2.1.40 Rocket engine, vehicle-launched missile.- "Vehicle-launched" operated prior to or during launching from an inhabited vehicle.

6.2.1.41 Rocket engine, remotely launched missile.- "Remotely launched" denotes a propulsion rocket engine for a missile other than that specified in paragraph 6.2.1.40.

6.2.1.42 Thrust.- Thrust (F) is the reactive force of the rocket engine during operation.

6.2.1.43 Thrust, effective.- The effective thrust is the effective impulse divided by the time interval between the two-90-percent-of-rated thrust points.

6.2.1.44 Thrust, mean.- The mean thrust is the total impulse divided by the duration.

6.2.1.45 Velocity, characteristic.- The characteristic velocity (c*) is the product of throat area in square inches, chamber pressure in pounds per square inch absolute, and nominal acceleration due to gravity (32.174) in feet per second divided by rate of propellant flow in pounds per second.

6.2.1.46 Effective duration.- The effective duration is the time interval between the two 90-percent-of-rated thrust points, as defined in paragraph 6.2.1.27 of Specification MIL-E-5150A.

6.2.1.47 Nominal chamber pressure.- The nominal chamber pressure is the nozzle stagnation pressure of the thrust chamber. Nozzle stagnation pressure is computed from a knowledge of measured values of injector end static pressure. Based upon presently used combustion and fluid flow theories, the ratio of injector end pressure to nozzle stagnation pressure is 1.068.

6.2.1.48 Engine performance test.- An engine performance test is a test of sufficient duration for the engine and recording equipment to reach steady state conditions for a period not less than 3 seconds and in which time all required performance measurements are obtained.

6.2.1.49 Critical components.- Critical components are the thrust chamber, turbopump, injector, main propellant and gas generator orifices, and gas generator assembly.

6.2.1.50 MD number identification.- MD number identification (end item markings) will be formulated and interpreted as shown in the following table:

END ITEM MARKINGS

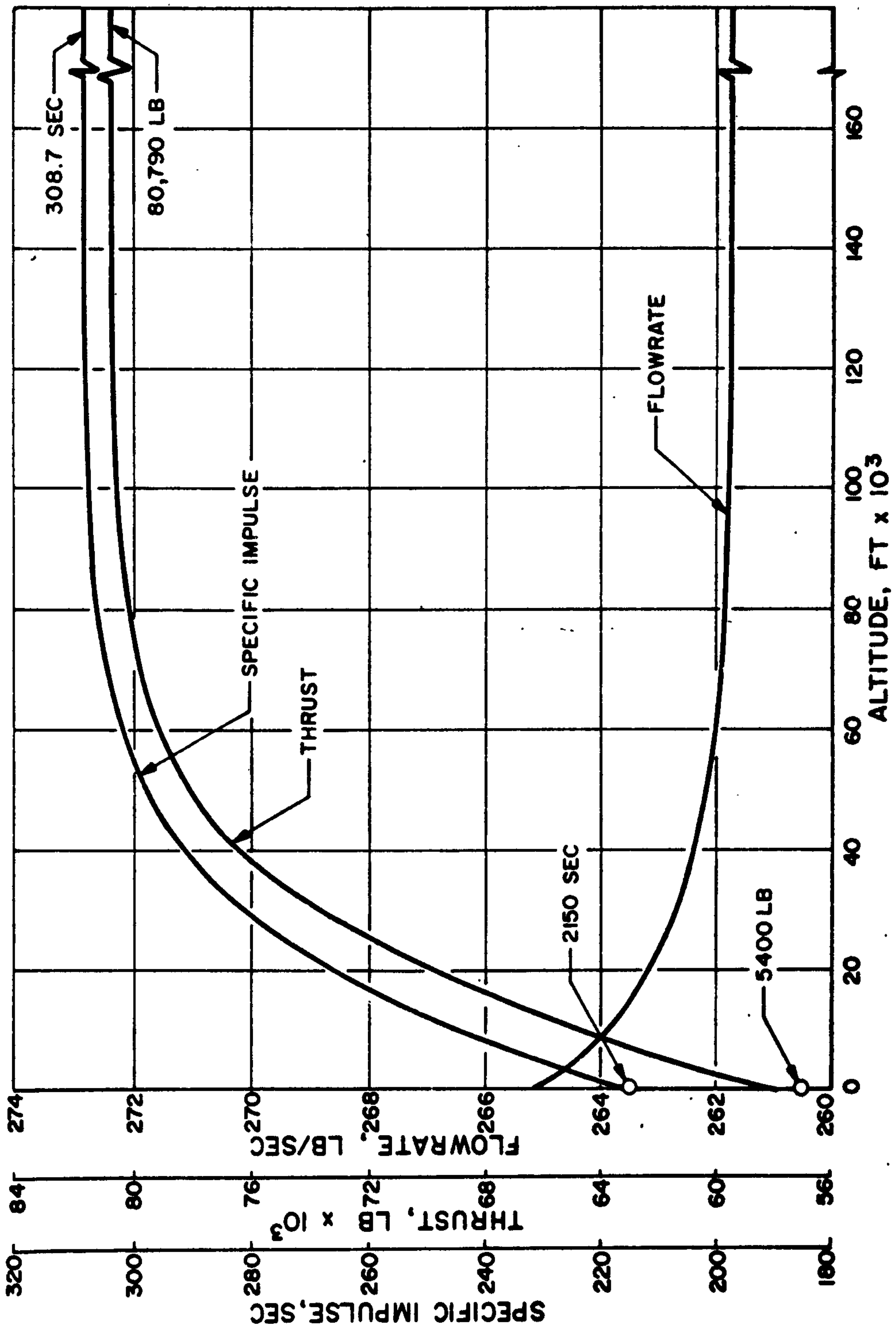
EXAMPLE	MD NUMBERED CHANGES INCORPORATED	ITEM MARKING
1	1, 2, 3, 4,	MD4
2	3,	MDx3
3	1, 2, 3, 5, 8	MD3x5x8
4	3, 4, 5, 7,	MDx35x7
5	1, 2, 3, 5, 6, 7, 8, 9, 10, 12	MD3x5<u>10</u>x<u>12</u>
6	EXAMPLE 4 AFTER INCORPORATION OF CHANGES 2 & 6	MDx27

6.2.2 Symbols.- Symbols used in this specification are defined as follows:

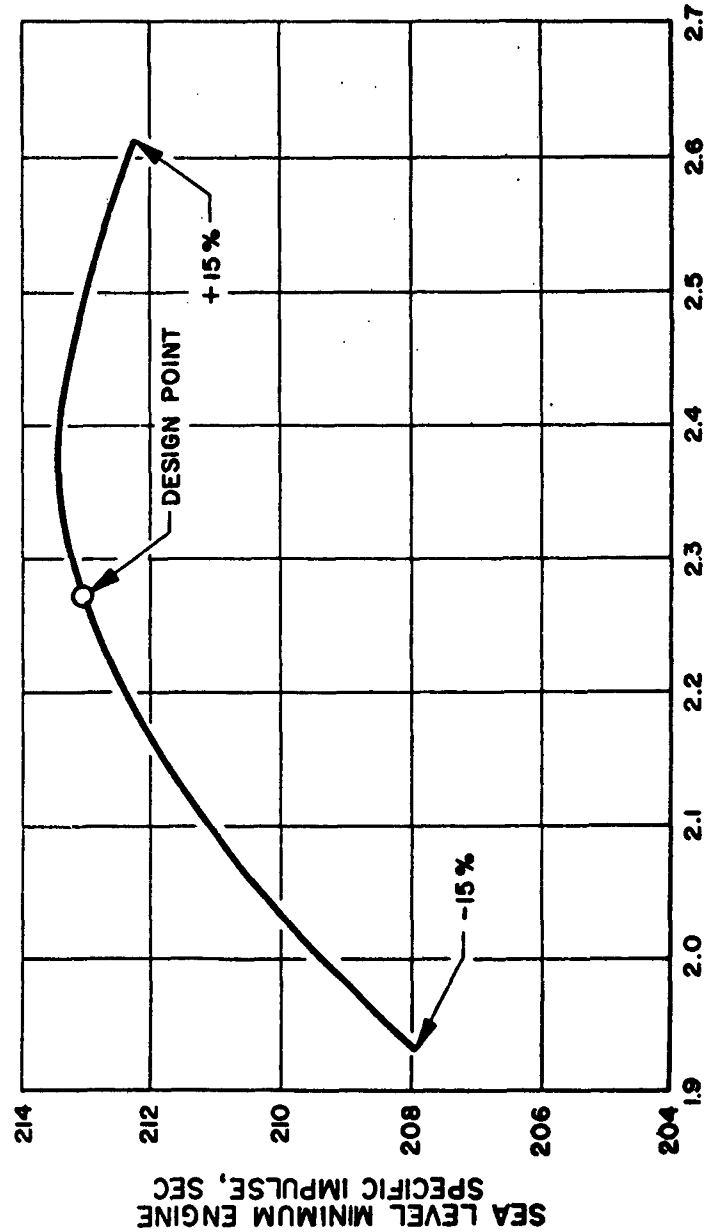
SYMBOL	QUANTITY	UNIT
c^*	Characteristic velocity	(ft/sec)
P_c	Chamber pressure	(psia)
A_t	Throat area	(in^2)
I_{sp}	Specific impulse	(sec)
$\dot{W}$	Fluid flow rate	(lb/sec)
C_F	Thrust coefficient	$\left(= \frac{F}{P_c A_t}\right)$
F	Thrust	(lbs)
L^*	Characteristic length	(in.) $\left(= \frac{V_c}{A_t}\right)$
V_c	Combustion chamber volume	(in^3)

6.3 Rocket engine mockup procedure.- Official examination of the mockup of new types of engine will be conducted in accordance with the provisions of ANA Bulletin No. 406.

6.4 Design and installation criteria.- Design criteria and recommended practices for the guidance of design shall be as set forth in ANA Bulletin No. 428.

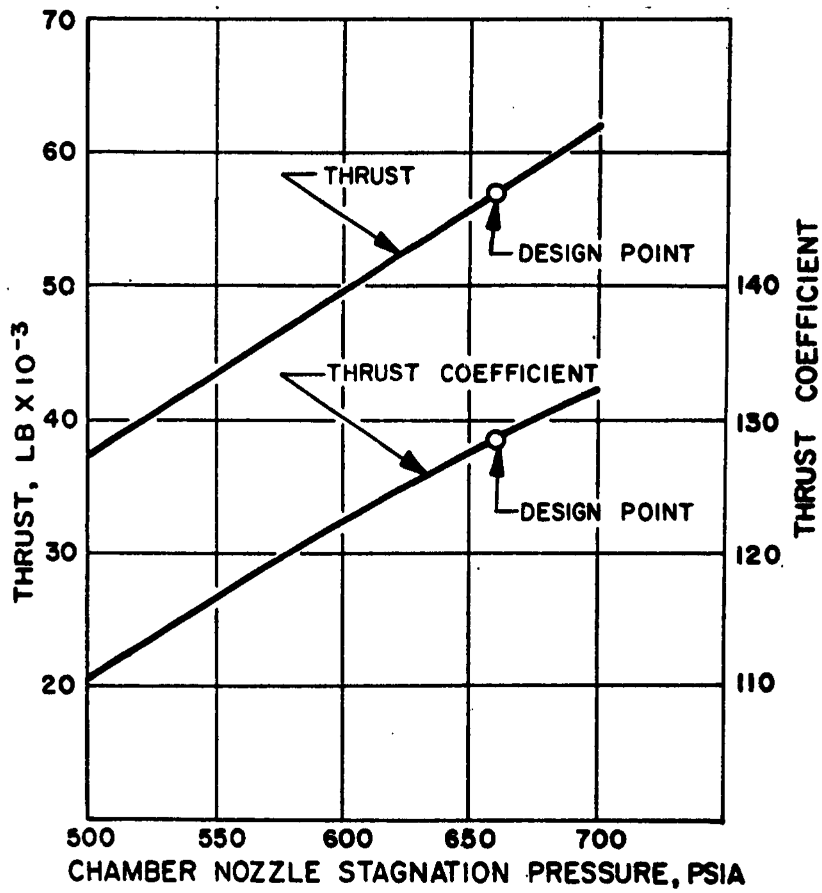


ESTIMATED NOMINAL ALTITUDE PERFORMANCE
YLR105-NA-7
FIGURE 1a



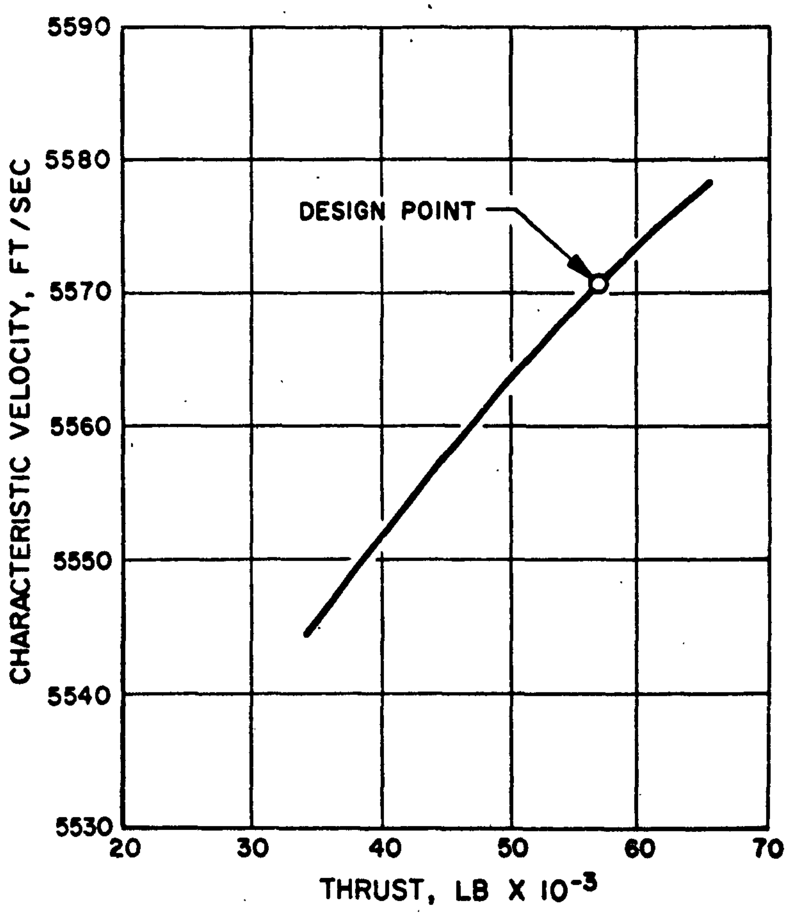
ESTIMATED MINIMUM ENGINE SPECIFIC IMPULSE VERSUS ENGINE MIXTURE RATIO
YLR105-NA-7

FIGURE 1b



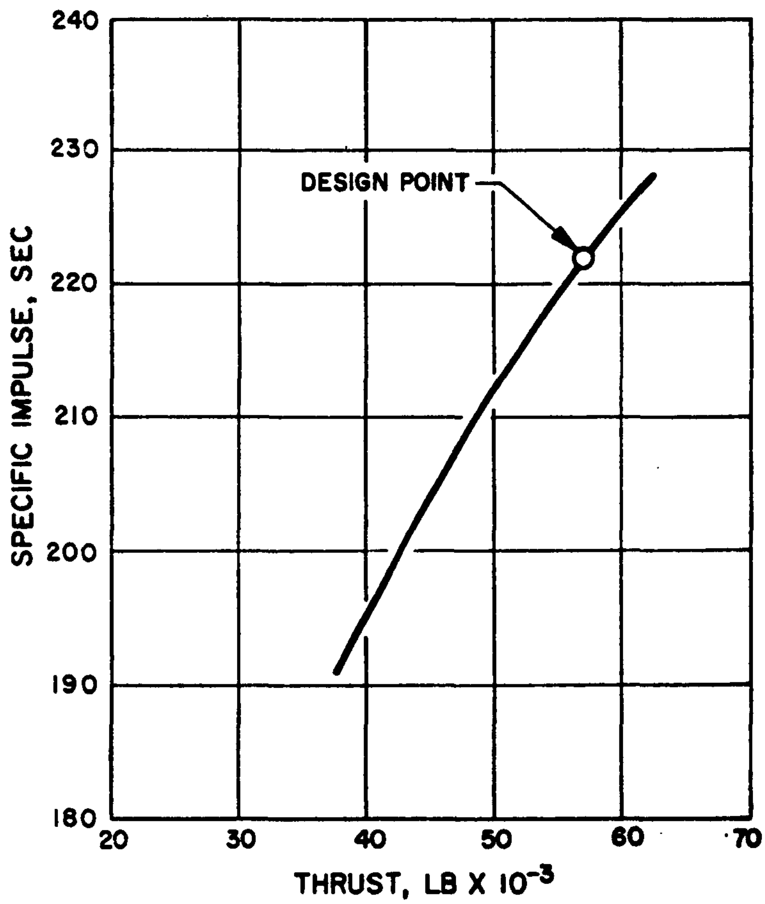
ESTIMATED SEA LEVEL THRUST & THRUST COEFFICIENT
VERSUS
CHAMBER NOZZLE STAGNATION PRESSURE
AT NOMINAL MIXTURE RATIO
YLR 105-NA-7 THRUST CHAMBER
(NOZZLE EXPANSION AREA RATIO, 25:1)

FIGURE 2



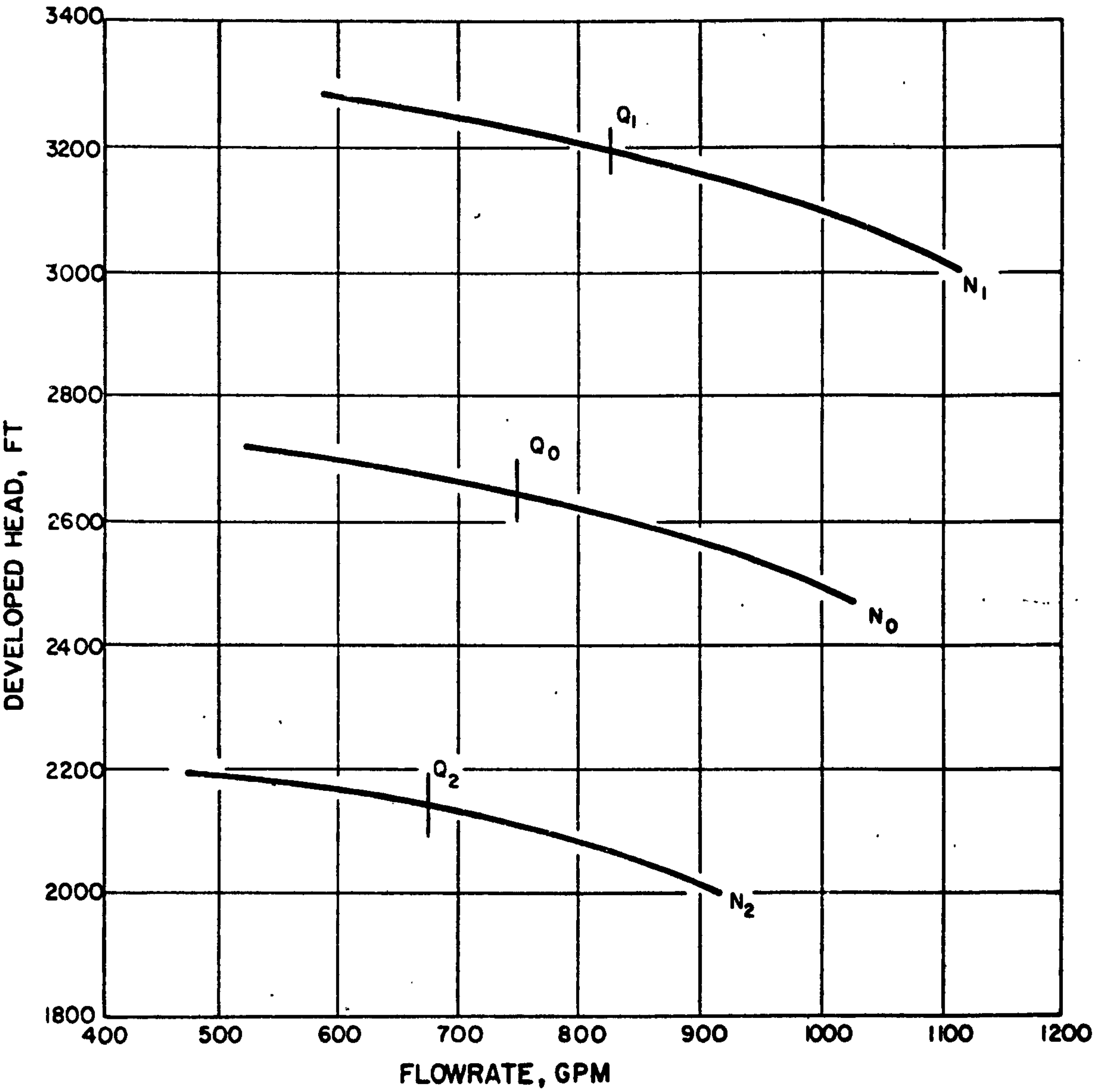
ESTIMATED CHARACTERISTIC VELOCITY
V.S.
SEA LEVEL THRUST AT NOMINAL MIXTURE RATIO
YLR105-NA-7 THRUST CHAMBER

FIGURE 3



ESTIMATED SEA-LEVEL SPECIFIC IMPULSE
VERSUS THRUST
AT NOMINAL MIXTURE RATIO
YLR105-NA-7 THRUST CHAMBER

FIGURE 4



N_0 = NOMINAL OPERATING SPEED	NOMINAL OPERATING SPEED 10156 RPM
N_1 = 10% ABOVE N_0	FLUID PP-1
N_2 = 10% BELOW N_0	FLUID VAPOR PRESSURE 01-3.0 PSI
Q_0 = NOMINAL FLOWRATE	FLUID TEMPERATURE 80F
Q_1 = 10% ABOVE Q_0	FLUID DENSITY 50.15 LB/FT^3
Q_2 = 10% BELOW Q_0	IMPELLER DIAMETER: 85 IN

DEVELOPED HEAD VS. VOLUMETRIC FLOWRATE AT CONSTANT SPEED
YLR 105-NA-7 FUEL PUMP
FIGURE 5a

DEVELOPED HEAD VS VOLUMETRIC FLOWRATE
AT CONSTANT SPEED, YLR105-NA-7 OXIDIZER PUMP

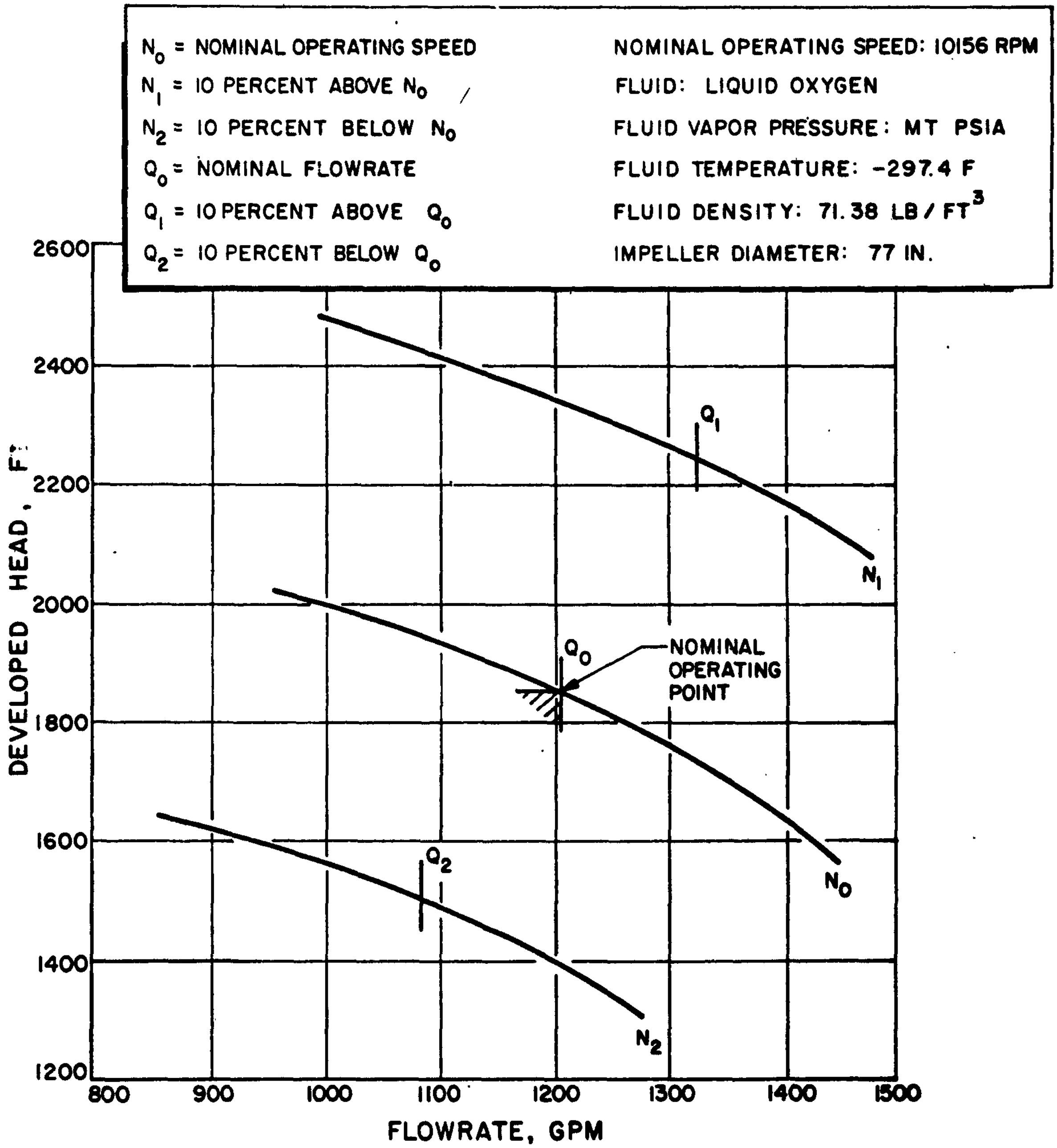
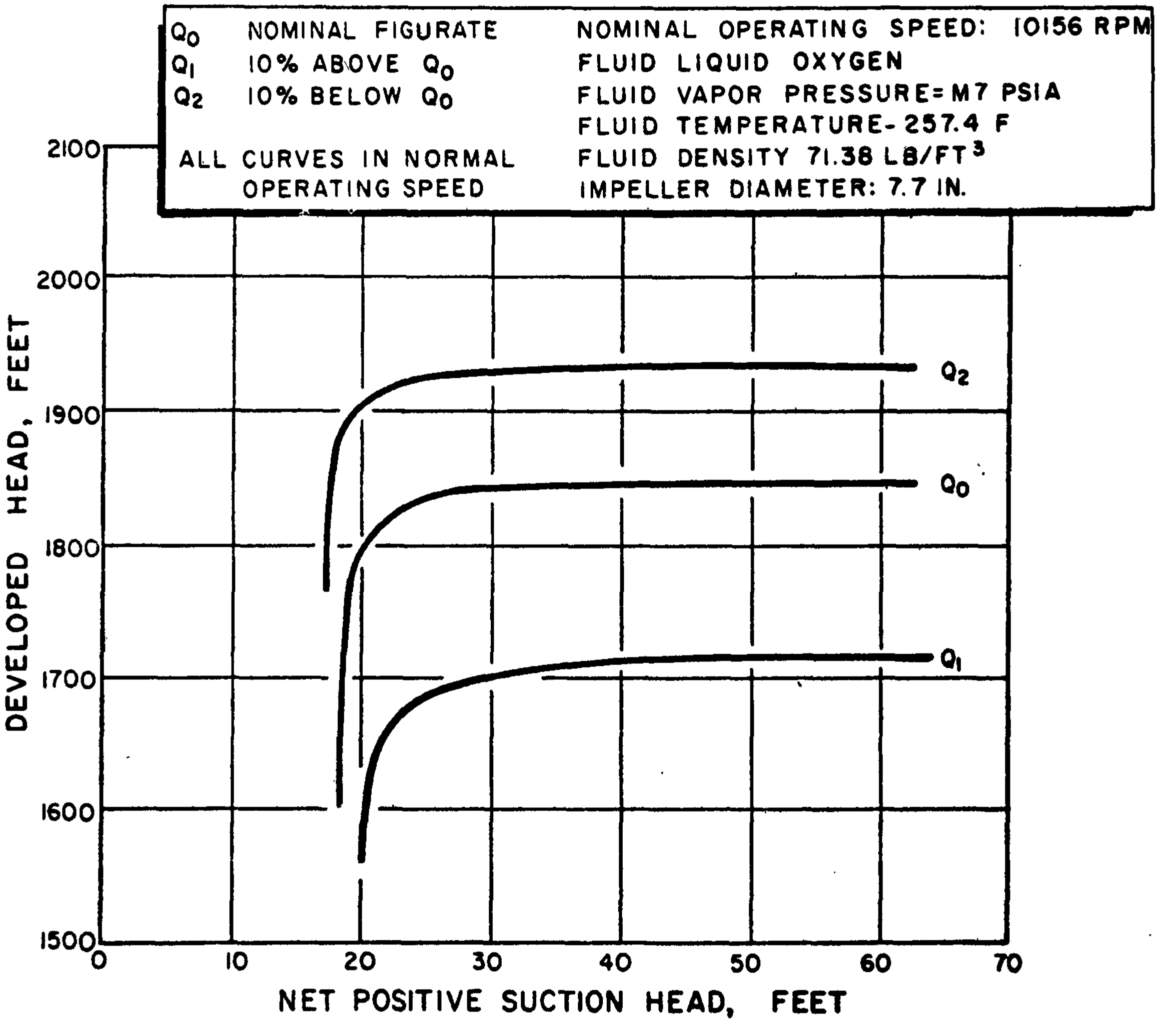


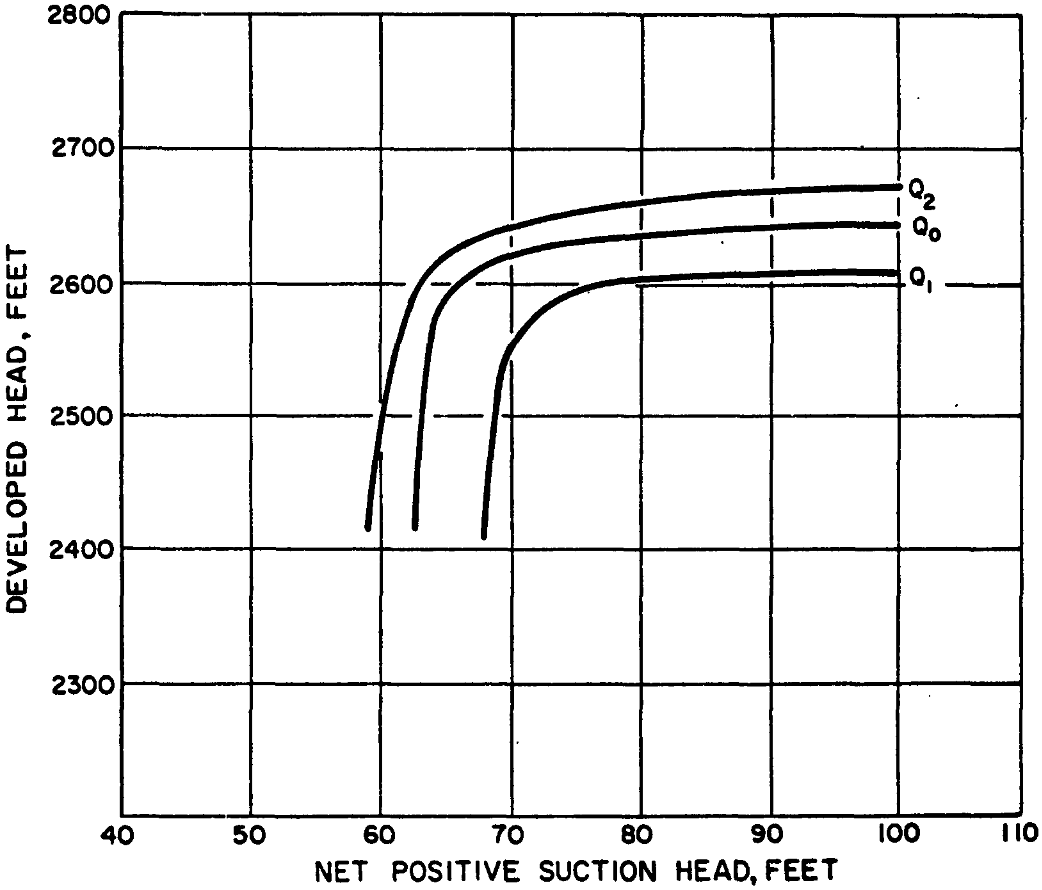
FIGURE 5b



CAVITATION CHARACTERISTICS AT CONSTANT SPEED (CAPACITY)

YLR105-NA-7 OXIDIZER PUMP

FIGURE 6a



Q_0 = NOMINAL FLOWRATE
Q_1 = 10 PERCENT ABOVE Q_0
Q_2 = 10 PERCENT BELOW Q_0
ALL CURVES AT NOMINAL OPERATING SPEED

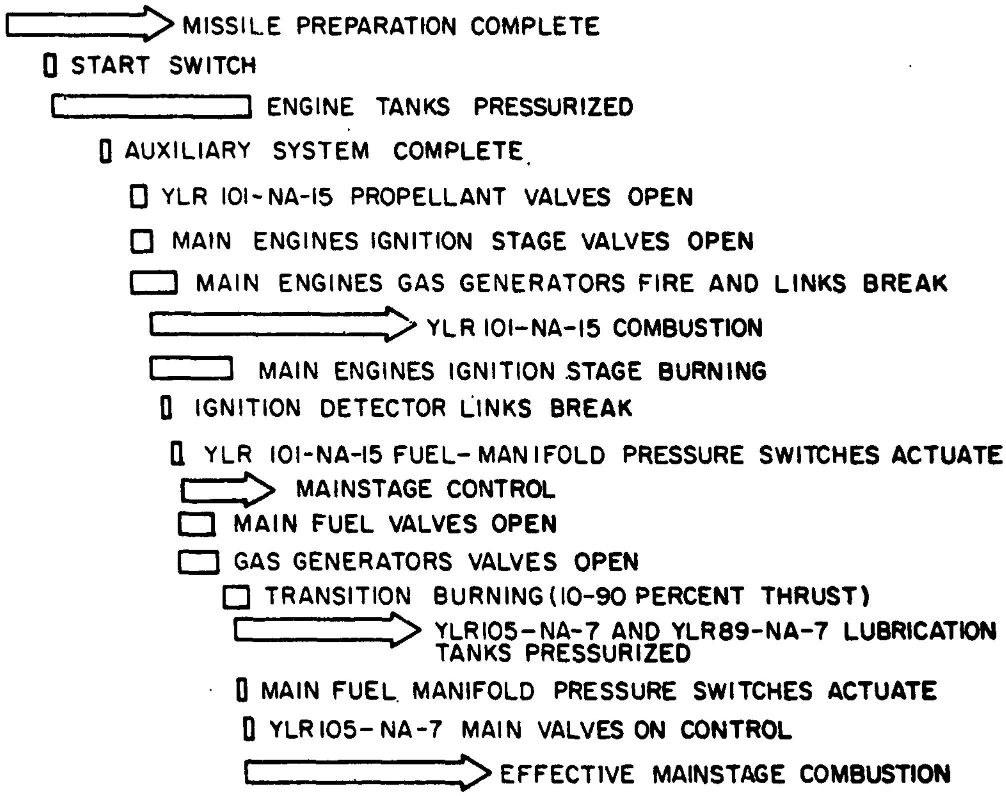
NOMINAL OPERATING SPEED 10156 RPM
FLUID RP-1
FLUID VAPOR PRESSURE: 01-30 PSIA
FLUID TEMPERATURE: 50 F
FLUID DENSITY: 50.45 LB/FT^3
IMPELLER DIAMETER = 85 IN.

CAVITATION CHARACTERISTICS AT CONSTANT SPEED AND CAPACITY

YLR105-NA-7 FUEL PUMP

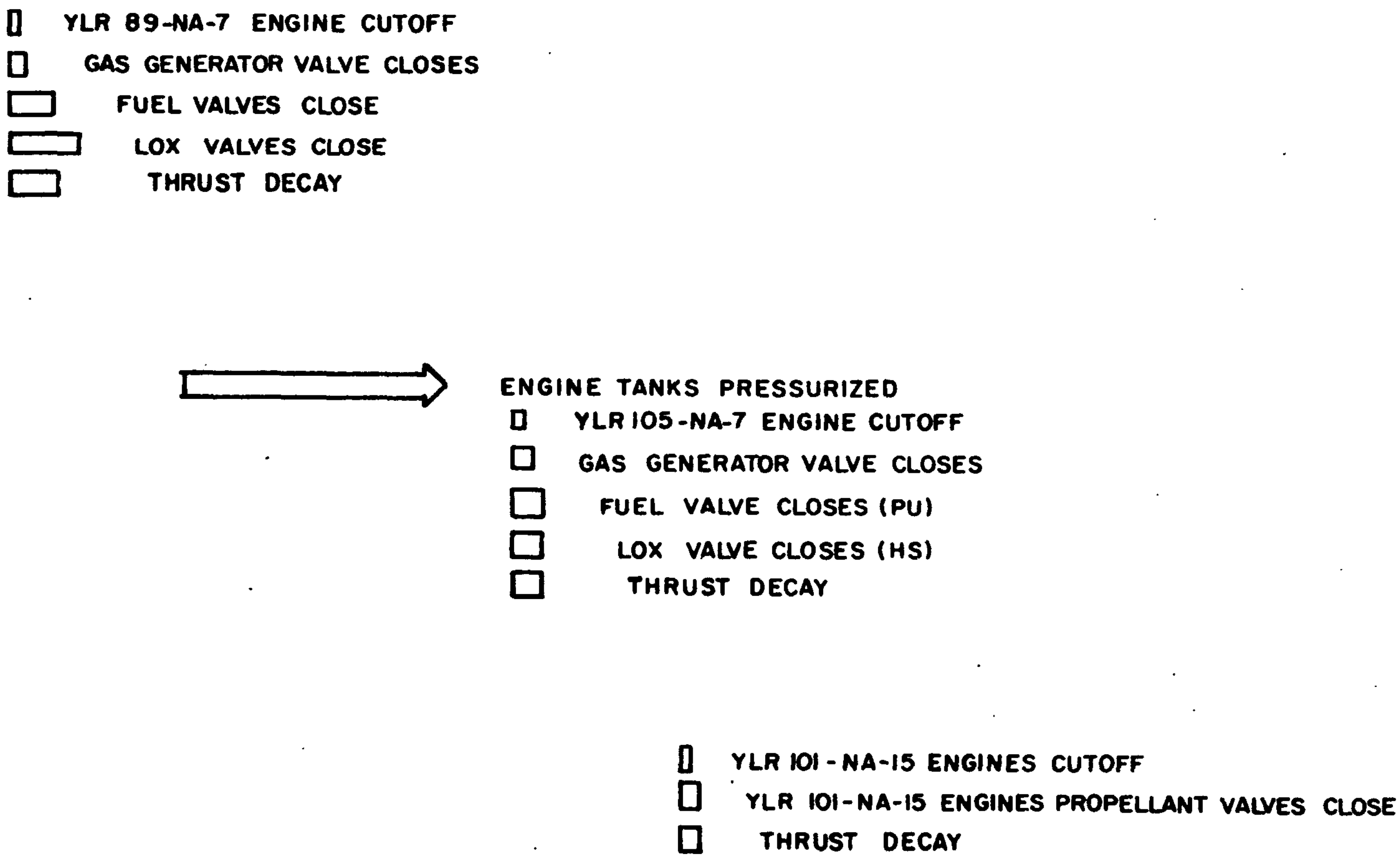
FIGURE 6b

NOTE:
MAIN ENGINES ARE CONSIDERED TO BE YLR89-NA-7 AND YLR105-NA-7



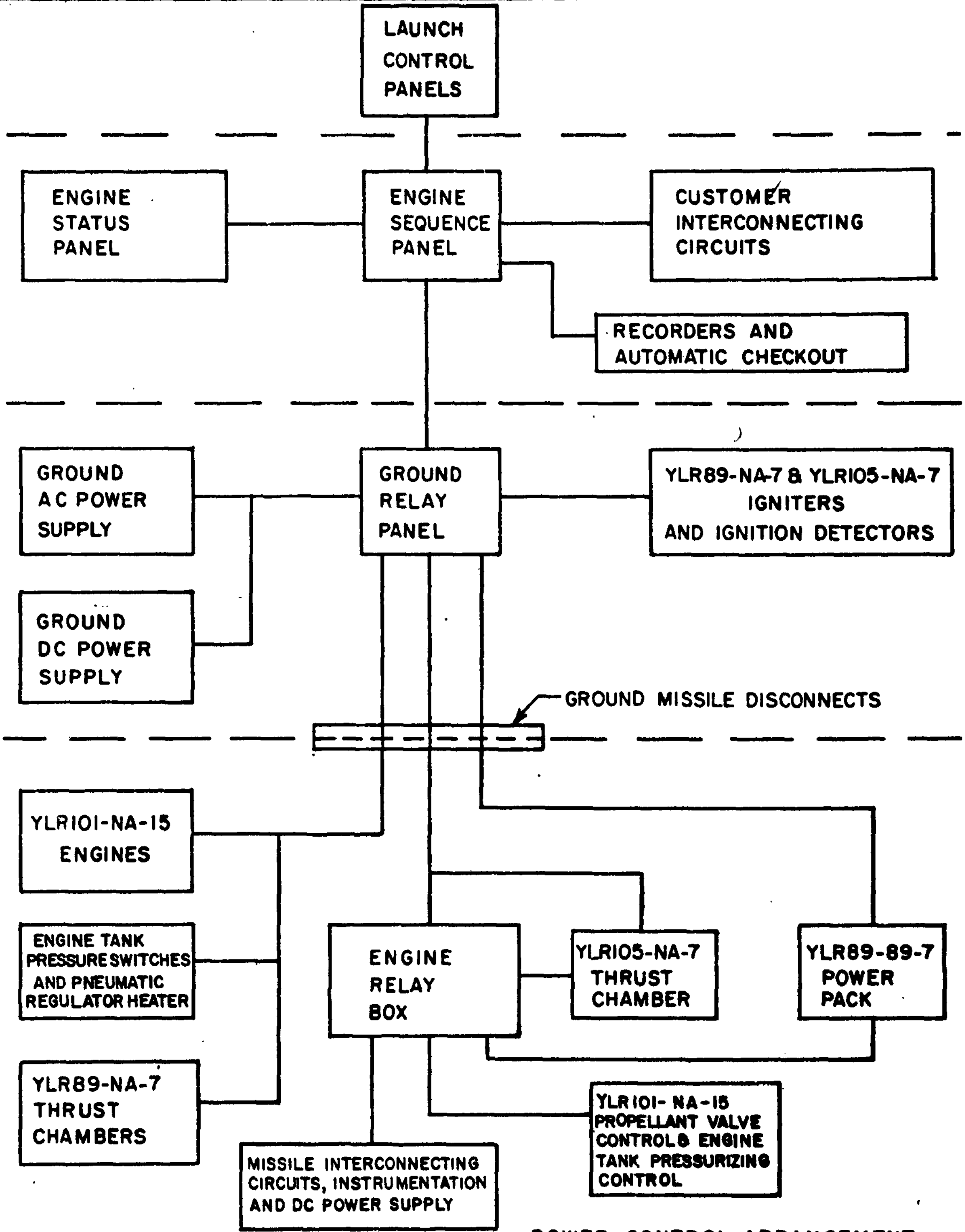
STARTING SEQUENCE
YLR89-NA-7, WHEN OPERATED WITH
YLR105-NA-7 AND YLR101-NA-15

FIGURE 7



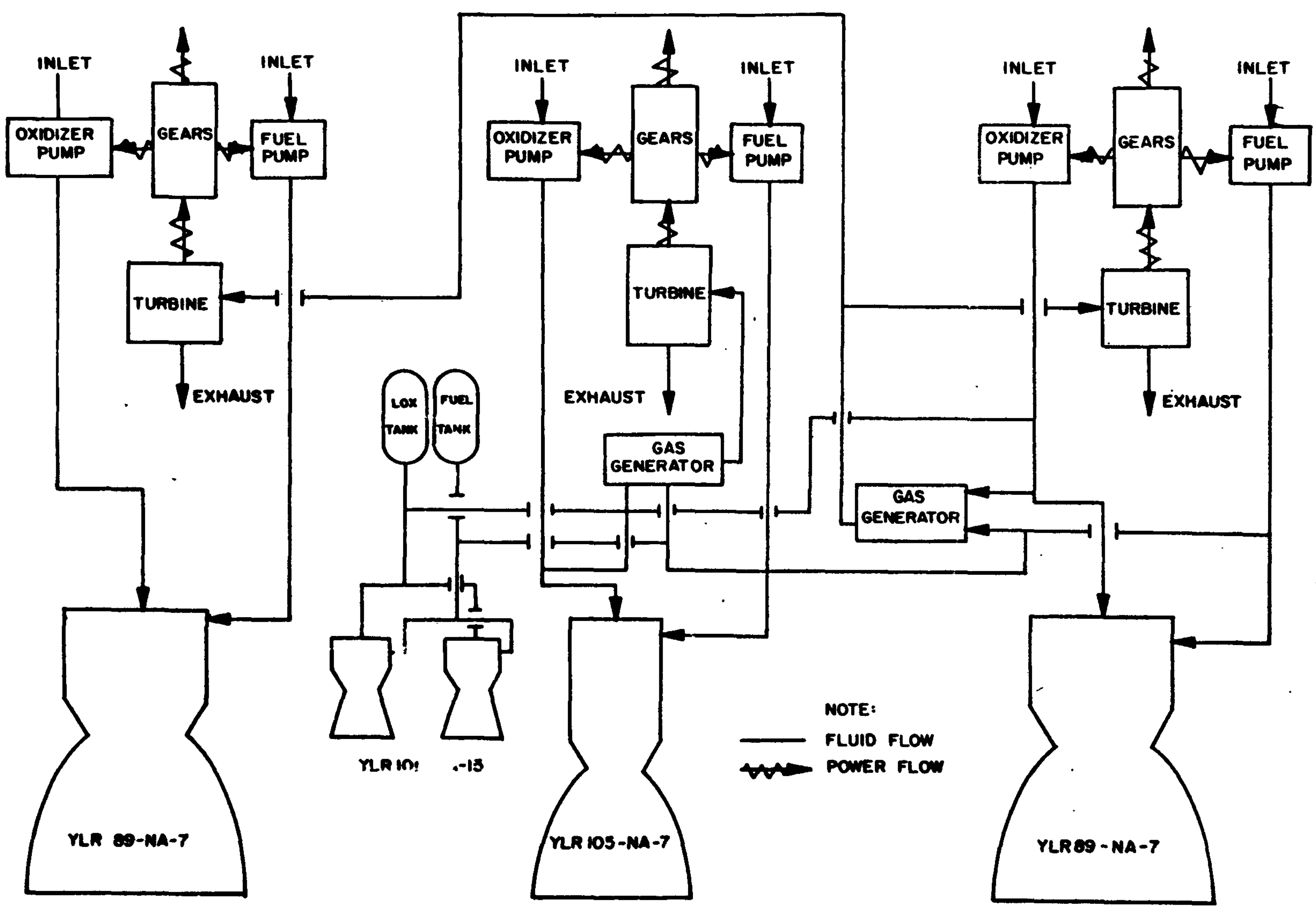
CUTOFF SEQUENCE

YLR105-NA-7, WHEN OPERATED WITH YLR89-NA-7 AND YLR101-NA-15

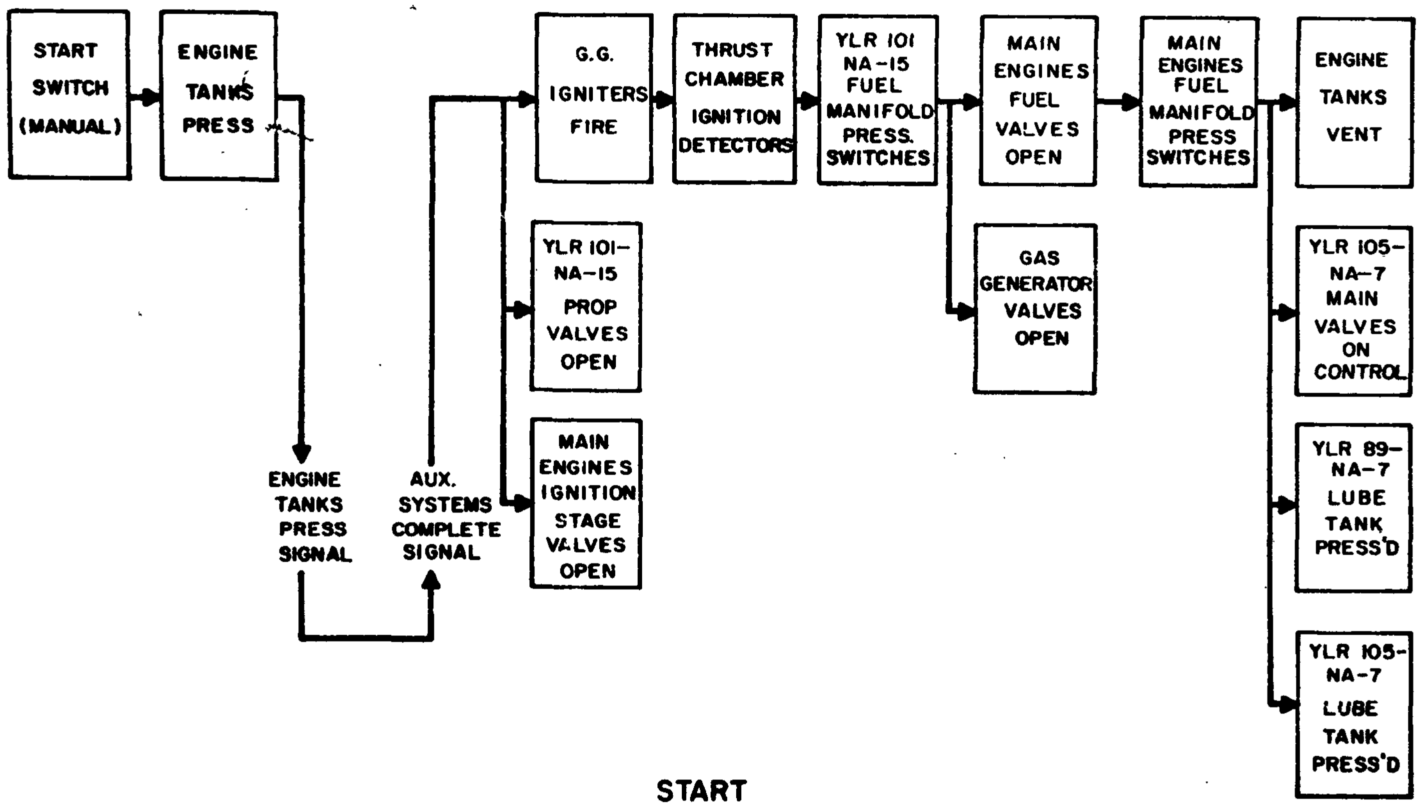


POWER CONTROL ARRANGEMENT, YLR105-NA-7, WHEN OPERATED WITH YLR89-NA-7 and YLR101-NA-15.

FIGURE 9

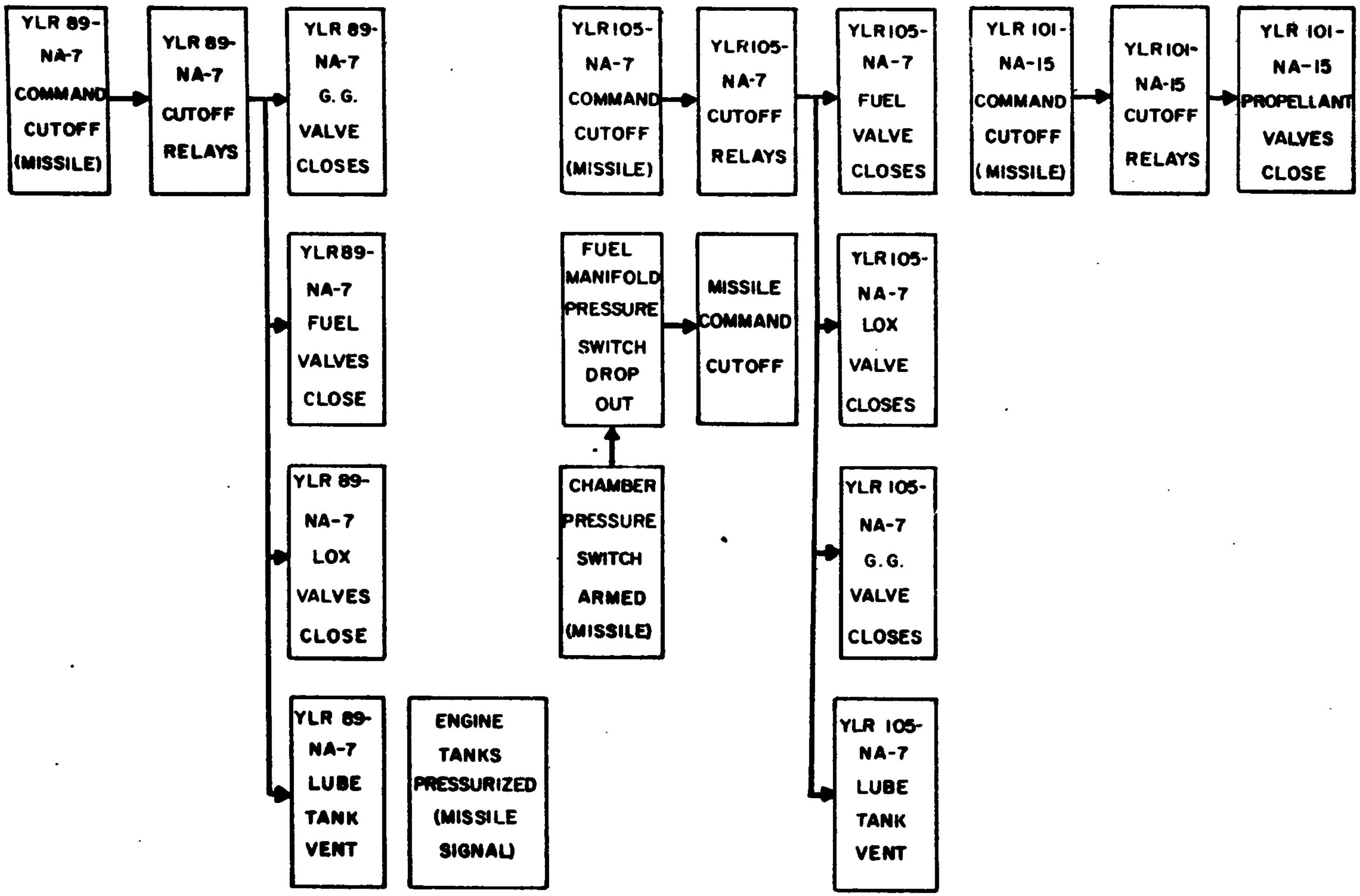


SCHEMATIC ROCKET ENGINE FLOW DIAGRAM
YLR105-NA-7, WHEN OPERATED WITH YLR89-7 AND YLR101-NA-15

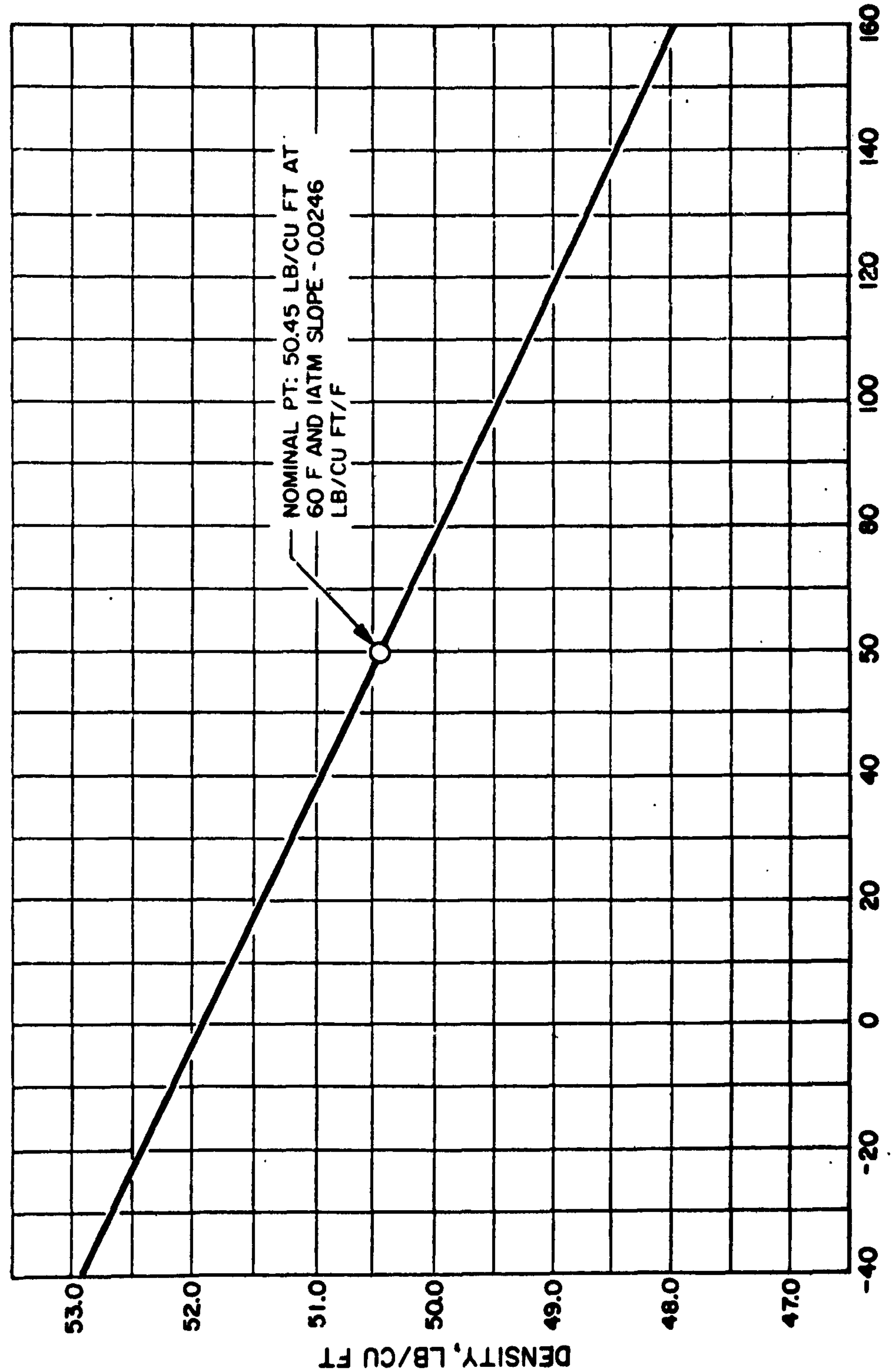


START
POWER CONTROL
INTERRELATION WITH ROCKET
ENGINE, FUNCTIONAL BLOCK DIAGRAM
OF YLR89-NA-7 WHEN OPERATED WITH YLR105-NA-7 AND YLR101-NA-15

FIGURE 11

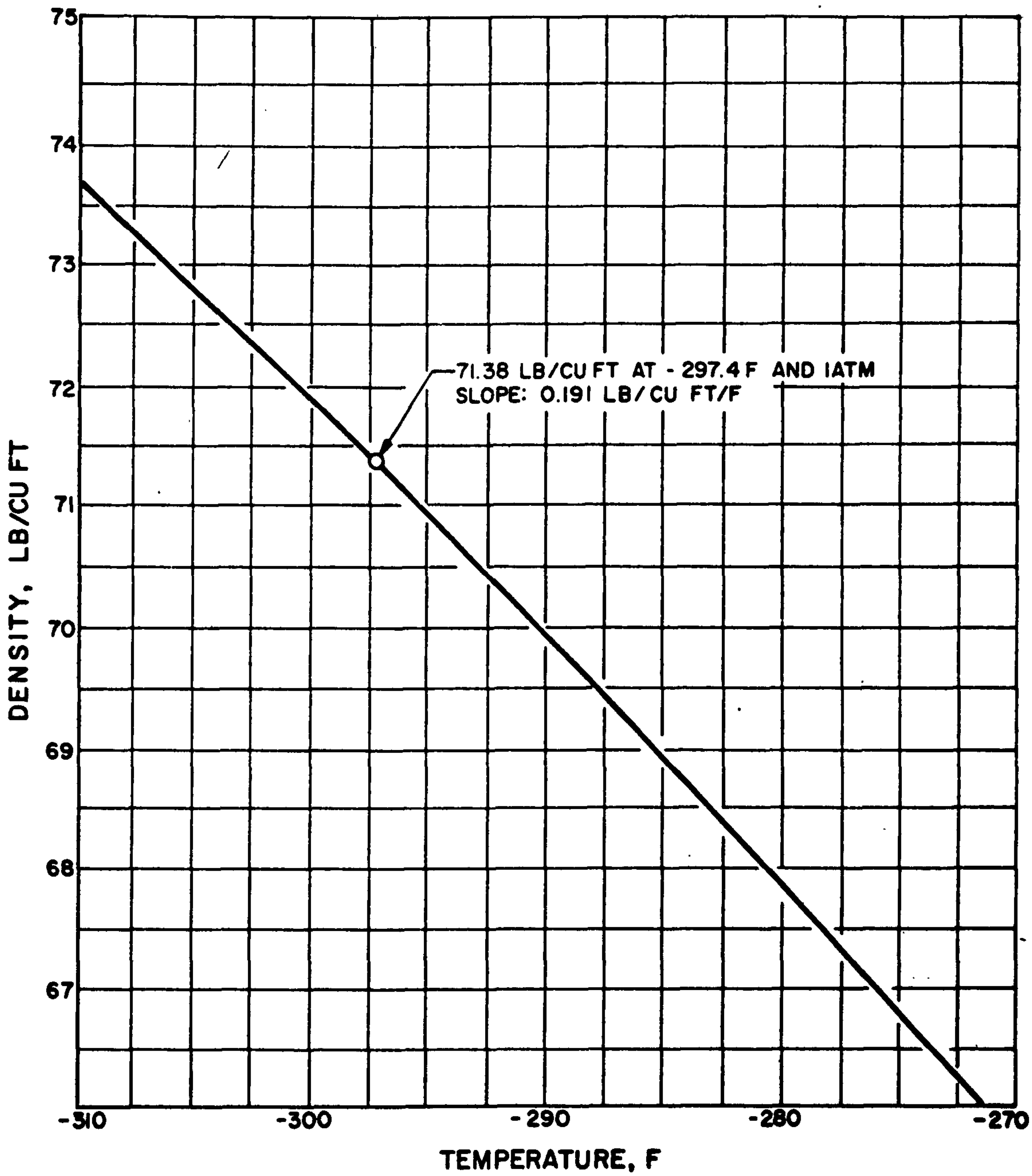


SHUTDOWN
POWER CONTROL
INTERRELATION WITH ROCKET
ENGINE, FUNTIONAL BLOCK DIAGRAM
OF YLR 89-NA-7 WHEN OPERATED WITH YLR 105-NA-7 AND YLR101-NA-15



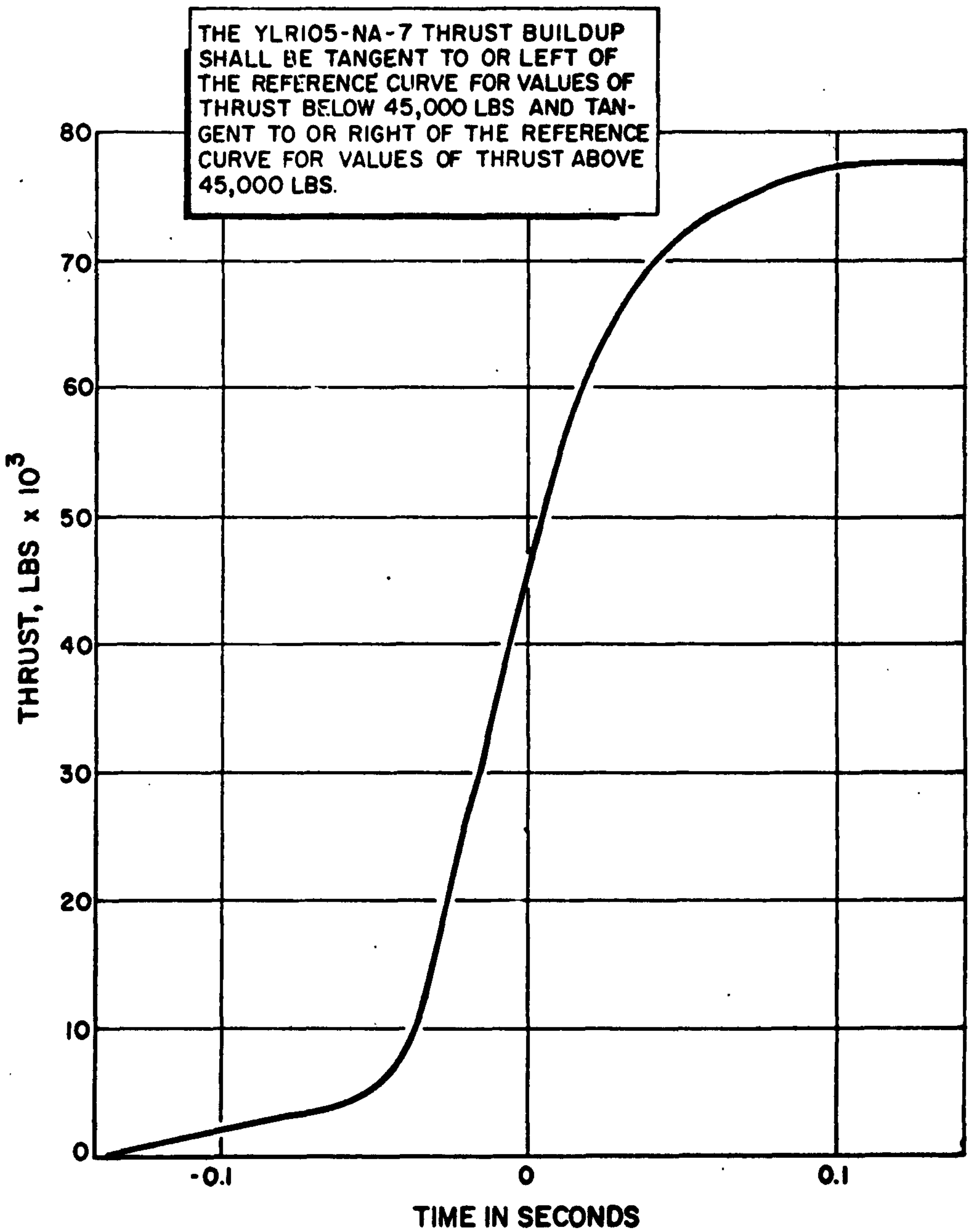
FUEL (RP-1) DENSITY
VS
TEMPERATURE

FIGURE 13



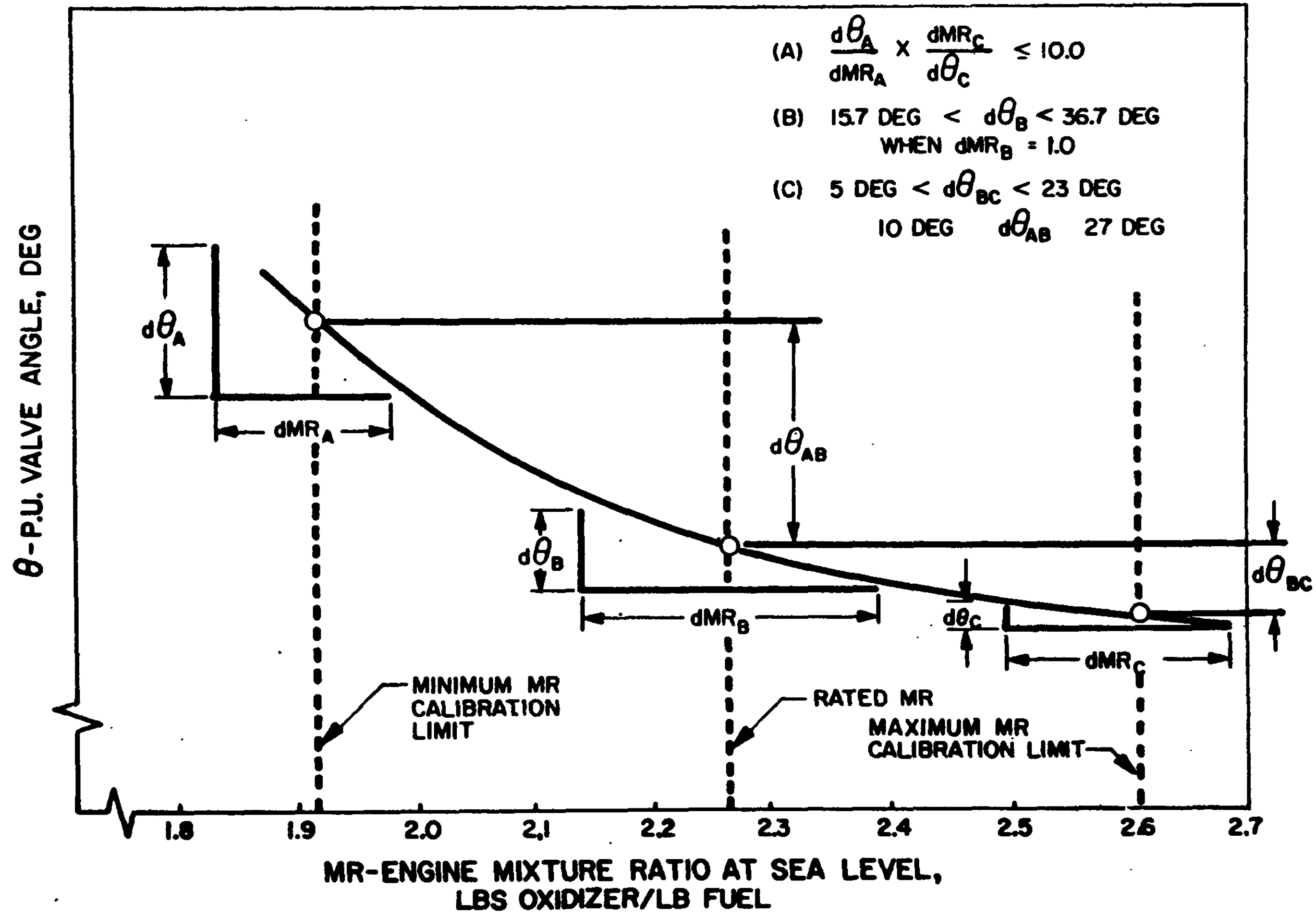
LIQUID OXYGEN DENSITY VS TEMPERATURE

FIGURE 14



YLR105-NA-7 THRUST BUILDUP SPECIFICATION
ALTITUDES: ZERO TO 10,000 FEET

FIGURE 15



LIMITS OF PROPELLANT-UTILIZATION VALVE ANGLE VS
ENGINE MIXTURE RATIO AT SEA-LEVEL STANDARD CONDITIONS

FIGURE 16